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## ABSTRACT

### INVESTIGATION OF ADVANCED PROPELLANTS TO ENABLE SINGLE STAGE TO ORBIT LAUNCH VEHICLES

Single-Stage-To-Orbit (SSTO) launch vehicles designs offer the promise of reduced complexity and cost compared to multi-stage vehicles, as only one stage needs be developed, produced, and maintained. Despite well-funded development efforts, no SSTO vehicles have been fielded to date. Existing chemical rocket and vehicle technologies do not enable feasible SSTO designs. In the future, new propellants with advanced properties could enable SSTO launch vehicles. A parametric sizing study was conducted to determine the rocket propulsion performance required. Bulk density, specific impulse, and main propulsion thrust-to-weight level were varied to determine the sensitivity of SSTO vehicle size to these parameters. Advanced propellants including strained-ring hydrocarbons and polynitrogen compounds are evaluated for their suitability in SSTO applications.

Jason Mossman  
26 July 2006



INVESTIGATION OF ADVANCED PROPELLANTS  
TO ENABLE SINGLE STAGE TO ORBIT  
LAUNCH VEHICLES

by  
Jason Mossman

A project  
submitted in partial  
fulfillment of the requirements for the degree of  
Master of Science in Mechanical Engineering  
in the College of Engineering  
California State University, Fresno  
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## TABLE OF CONTENTS

	Page
LIST OF TABLES . . . . .	vii
LIST OF FIGURES . . . . .	viii
Chapter	
1. INTRODUCTION . . . . .	1
The Tsiolkovsky Rocket Equation. . . . .	3
History of SSTO Launch Vehicle Programs . . . . .	6
Current Interest in SSTO Launch Vehicles . . . . .	10
2. REVIEW OF EXISTING PARAMETRIC SSTO ROCKET SIZING STUDIES . . . . .	12
3. ADVANCED PROPELLANTS . . . . .	17
Propellant Performance. . . . .	17
High Energy Density Materials. . . . .	20
4. PARAMETRIC SSTO VEHICLE SIZING METHODOLOGY. . . . .	24
Study Parameters and Design of Experiments. . . . .	24
Mission and Operations Concept . . . . .	27
Weight Estimation Techniques. . . . .	30
Vehicle Sizing Techniques . . . . .	37
5. SSTO LAUNCH VEHICLE SIZE TRENDS . . . . .	41
Results for Near Term Structures with Minimal Insulation.. . . .	41
Results for Near Term Structures with Heavy Insulation. . . . .	44
Results for Advanced Structures with Minimal Insulation . . . . .	48
Results for Advanced Structures with Heavy Insulation. . . . .	52

6. CONCLUSIONS . . . . .	56
Feasibility of SSTO with LOX/LH <sub>2</sub> . . . . .	56
Feasibility of SSTO with Advanced Propellants . . . . .	57
SSTO vs. TSTO for Economical RLVs . . . . .	58
REFERENCES . . . . .	59
APPENDICES . . . . .	65
A. EXAMPLE SSTO VEHICLE MASS ESTIMATE . . . . .	66
B. EXAMPLE POST INPUT FILE. . . . .	70
C. EXAMPLE TRAJECTORY OUTPUT DATA . . . . .	75

## LIST OF TABLES

Table	Page
1. Properties of Potential Launch Vehicle Propellants . . . . .	5
2. Properties of Representative Strained Ring Hydrocarbon Propellants .	22
3. Oxidizer and Fuel Densities Selected for O:F of 6.0. . . . .	25

## LIST OF FIGURES

Figure	Page
1. S STO and T STO Gross Lift-Off Weight Estimates . . . . .	4
2. Common Nozzle Geometries (Sutton and Biblarz) . . . . .	8
3. XRS-2200 Linear Aerospike Rocket Engine (Kotake et al.). . . . .	8
4. Common Rocket S STO Configurations (Cook) . . . . .	9
5. Orbital Elements (Bate, Mueller and White) . . . . .	13
6. AMLS Dry Weight Sensitivity to I <sub>sp</sub> and T/W (Stanley et al.) . . . .	15
7. Theoretical I <sub>sp</sub> as a function of M and T <sub>0</sub> . . . . .	18
8. Gross Weight for S STO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 150:1. . . . .	41
9. Gross Weight for S STO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 100:1. . . . .	42
10. Gross Weight for S STO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 50:1 . . . . .	42
11. Dry Mass for S STO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 150:1. . . . .	43
12. Dry Mass for S STO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 100:1. . . . .	43
13. Dry Weight for S STO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 50:1 . . . . .	44
14. Gross Weight for S STO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 150:1. . . . .	45
15. Gross Weight for S STO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 100:1. . . . .	45
16. Gross Weight for S STO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 50:1 . . . . .	46
17. Dry Weight for S STO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 150:1. . . . .	46

Figure	Page
18. Dry Weight for SSTO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 100:1. . . . .	47
19. Dry Weight for SSTO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 50:1 . . . . .	47
20. Gross Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 150:1. . . . .	49
21. Gross Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 100:1. . . . .	49
22. Gross Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 50:1 . . . . .	50
23. Dry Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 150:1. . . . .	50
24. Dry Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 100:1. . . . .	51
25. Dry Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 50:1 . . . . .	51
26. Gross Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 150:1. . . . .	52
27. Gross Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 100:1. . . . .	53
28. Gross Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 50:1 . . . . .	53
29. Dry Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 150:1. . . . .	54
30. Dry Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 100:1. . . . .	54
31. Dry Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 50:1 . . . . .	55
32. Region of SSTO Designs with GLOW less than 1.5 Mlb <sub>m</sub> . . . . .	57

## Chapter 1

### INTRODUCTION

The high cost of access to space is widely regarded as the most significant barrier preventing further exploration and economic development beyond the Earth.<sup>1</sup> These costs vary based on the specifics of the mission and the launch vehicle chosen, but typical values range from \$2500 to in excess of \$20,000 *per pound* placed in Low Earth Orbit (LEO).<sup>2</sup> Many launch vehicle architectures have been proposed that offer to reduce these costs.

The vast majority of launch vehicles have been completely expendable, with no attempt at recovery or reuse of the components. If relatively few flights are required, this is the least expensive solution, as the increased expense of developing, building, and maintaining a reusable system would exceed the cost of the expended hardware.<sup>3</sup> As the number of flights required increases, recovery and reuse of some or all of the launch vehicle becomes economically viable. The partially reusable Space Transportation System (STS), or Space Shuttle, is the most notable example of an attempt to achieve cost savings through the reuse of hardware.

Unfortunately, the costs associated with reuse of the STS have proven to be much higher than was anticipated during that system's development. The FY06 budget for the system was \$4.47 billion,<sup>4</sup> yet at the time of this writing, the vehicle has flown only once and is scheduled for only one more flight during the fiscal year.<sup>5</sup> In fact, during no year has the system been used more than nine times, despite projections during development that 55 annual flights would be required to justify reuse.<sup>2,6</sup> The STS has not achieved the desired flight rate because the

recovery, refurbishment, and reassembly of the vehicle are more complex and time-consuming than originally planned. While early plans envisioned 160 hours to refurbish, assemble, and launch the vehicle, actual timelines have been on the order of 3 months prior to the loss of the *Challenger*, 6 months after that incident, and the timelines have been extended even further following the loss of *Colombia*.<sup>2,5,7</sup> In retrospect, it is not surprising that the system often referred to as “the most complex machine ever built”<sup>8</sup> can not be refurbished in less than a week, especially considering the high consequences of failure if mistakes are made.

Recognizing the potential for reducing cost by reducing complexity, the Single-Stage-To-Orbit vehicle has been identified by many studies as a way to reduce system complexity by eliminating stages.<sup>9</sup> Much of the rationale for developing a fully reusable SSTO vehicle evolved from experience with the partially reusable, multi-stage STS, and the high costs associated with the many elements and associated infrastructure required for that system. The STS consists of the Orbiter, the External Tank (ET), and a pair of Solid Rocket Boosters (SRBs). Each element of the vehicle has its own unique infrastructure, equipment, and support personnel, and then the elements must be assembled on top of the Mobile Launch Platform in the massive Vehicle Assembly Building. For the STS, the SRBs must be retrieved from the Atlantic Ocean, then shipped back to Utah for refurbishment and returned to Kennedy Space Center in Florida, while the ET arrives on a barge from its manufacturing site in Louisiana. Many of the subsystems in the Orbiter are removed for maintenance in the Orbiter Processing Facility.



### The Tsiolkovsky Rocket Equation

While there may be operational advantages to an SSTO launch vehicle, there is a key disadvantage to the SSTO approach that has prevented the development of a successful SSTO vehicle: SSTO vehicles require very efficient engines and structures relative to those required for multi-stage vehicles. The maximum change in velocity that any single rocket stage can achieve is given by the Tsiolkovsky rocket equation<sup>10</sup>

$$\Delta V = I_{sp} \cdot g_0 \cdot \left( \frac{m_{initial}}{m_{final}} \right)$$

where  $I_{sp}$  is the vacuum specific impulse,  $g_0$  is acceleration due to gravity, and  $m_{initial}$  and  $m_{final}$  represent the initial and final masses of the stage, respectively. Specific impulse is a key metric for the performance of rocket engines, and is defined as the thrust of the engine divided by the weight flow rate of propellants through the engine.<sup>11</sup> In the United States, specific impulse is usually measured in units of  $\text{lb}_f$  per  $\text{lb}_m/\text{sec}$ , referred to as “seconds.” The ratio of initial mass to final mass is referred to as the mass ratio, and is a measure of the structural efficiency of a stage. Another parameter used to measure structural efficiency of a rocket stage is the propellant mass fraction, the ratio of propellant mass to gross mass of a stage, denoted by  $\zeta$ . Most rocket stages have propellant mass fractions in the range of 0.85 to 0.95.<sup>11</sup> Stages with low density propellants tend to have lower propellant mass fraction due to larger propellant tanks. Reusable stages will also tend towards lower propellant mass fraction than expendable stages due to the addition of numerous subsystems (wings, thermal protection material, etc.) that may be required to recover a stage.

Using the rocket equation and assuming various values for propellant mass fraction, the Gross Lift-Off Weight (GLOW) of both SSTO and Two Stage To Orbit (TSTO) vehicles can be roughly estimated without difficulty. Figure 1

presents a graph of such estimates as a function of specific impulse for propellant mass fractions of 0.88, 0.89, and 0.90, assuming that 30,000 feet/sec [9.1 km/sec] of delta-V is required for the mission, that the payload to be delivered weighs 10,000 pounds [4536 kg], and that 34% of the delta-V is delivered by the first stage of the TSTO system. These assumptions are consistent with typical conceptual design studies.<sup>11,12</sup> Several trends are apparent from this graph. First, the gross weights are lower for the TSTO system. Second, for each SSTD concept, there is a value of specific impulse at which the gross weight becomes extraordinarily sensitive to perturbations in specific impulse. The magnitude of change in a vehicle's size to a small perturbation in vehicle dry weight is known as *growth factor*.<sup>12,13</sup> The growth factor is higher for the SSTD designs than for the TSTO designs and SSTD designs are also more sensitive to changes in specific impulse.

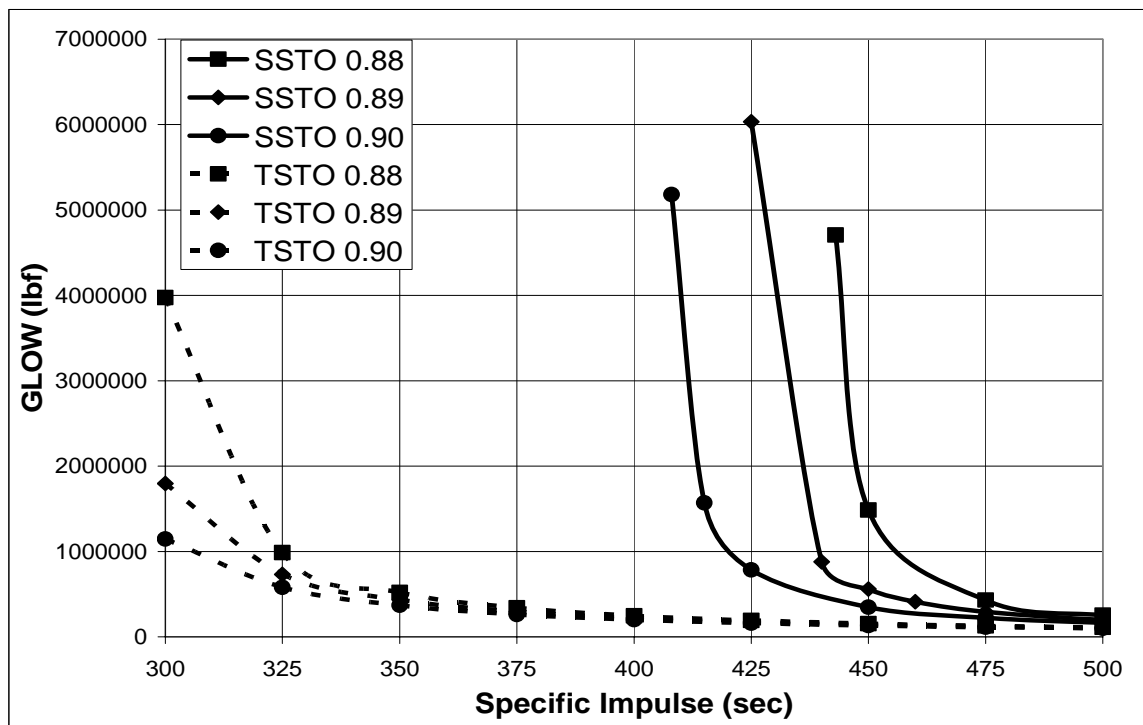


Figure 1. SSTD and TSTO Gross Lift-Off Weight Estimates

As a vehicle design matures from the conceptual phase towards preliminary design and into production of a prototype, the structural weight tends to increase and the engine performance tends to decrease. Due to the lower growth factor on TSTO vehicles, development of an SSTO vehicle will, in general, be at greater risk of cost or schedule overrun (or outright program failure) as the concept matures and vehicle weights and performance levels change.

Another trend visible in Figure 1 is that SSTO vehicles are not feasible for the propellant mass fractions shown if specific impulse is below 400 seconds, and growth factor would become much more manageable if it were 450 seconds or higher. For potential liquid rocket engine propellants, Table 1 provides the oxidizer density, fuel density,<sup>14</sup> oxidizer to fuel mass ratio (O:F), resultant bulk density of the overall propellant combination, and the vacuum specific impulse of the propellant combination as calculated using the Chemical Equilibrium for Applications program,<sup>15</sup> assuming a chamber pressure of 3750 psia [25.9 MPa], a nozzle expansion ratio of 70:1 and using typical values for combustion and nozzle efficiencies.

Table 1. Properties of Potential Launch Vehicle Propellants

Oxidizer	Fuel	Oxidizer Density (lb/ft <sup>3</sup> )	Fuel Density (lb/ft <sup>3</sup> )	O:F Ratio	Bulk Density (lb/ft <sup>3</sup> )	Vacuum I <sub>sp</sub> (sec)
LOX	RP-1	71.2	50.3	2.7	64.0	348.0
LOX	CH <sub>4</sub>	71.2	26.4	3.5	51.7	358.8
LOX	H <sub>2</sub>	71.2	4.4	6.0	22.5	451.7
F <sub>2</sub>	H <sub>2</sub>	93.6	4.4	10.0	32.9	477.1

Two propellants in Table 1 have a specific impulse greater than 450 seconds; liquid oxygen (LOX) with liquid hydrogen, and liquid fluorine with liquid hydrogen. Despite the excellent performance of fluorine, it has proven to be far too toxic to make a practical oxidizer and will not be considered further.<sup>16</sup> The remaining propellant combination, LOX and hydrogen, has the highest specific impulse of any chemical propellants in common use, and is currently used in many launch vehicles, including the STS and both of the Evolved Expendable Launch Vehicle designs used by the U.S. Air Force and NASA, the Delta IV and the Atlas V. However, the extremely low density of liquid hydrogen brings the bulk density of the LOX/hydrogen combination down to the point where the roughly 0.89 mass fraction shown to be required in Figure 1 for a low-growth factor SSTO will be challenging to achieve due to the enormous size of the hydrogen tank. The extremely low boiling point (-414° F at 50 psia) of hydrogen<sup>17</sup> requires the use of heavy insulation on the hydrogen tank which further reduces the propellant mass fraction. Using a hydrocarbon such as methane or RP-1 (a grade of kerosene) as fuel can significantly improve the bulk density, but it is clear from Figure 1 that the propellant mass fraction would have to significantly exceed 0.90 for these propellants to be used in an SSTO design.

#### History of SSTO Launch Vehicle Programs

The U.S. Air Force and NASA have been studying SSTO launch vehicles in one form or another since the 1960s.<sup>18</sup> The Air Force application of this technology was focused on a “military spaceplane” while NASA began studying follow-on vehicles to the STS before that system’s first launch. Among the first well-funded SSTO efforts was the National AeroSpace Plane (NASP) program, which ran from 1986 to 1995 and would have been a \$15 billion program had it

run to completion. The NASP vehicle design included an air-breathing supersonic combustion ramjet, or scramjet. As the vehicle would not need to use internally stored oxygen over a significant portion of its trajectory, the effective specific impulse of such a propulsion system can be significantly higher than that of pure rocket systems, though there are also several technical challenges associated with this type of vehicle that do not exist on pure rocket systems.<sup>19,20,21</sup> A thorough treatment of the advantages and disadvantages of scramjets in the use of space launch vehicles is beyond the scope of this project. Budget cuts following the end of the Cold War and low technology readiness levels in key areas led to the eventual cancellation of the NASP program.

In the early 1990s, the Strategic Defense Initiative Office ran the DC-X program, in which a subscale demonstrator successfully demonstrated vertical takeoff / vertical landing with a rocket, as well as the ability to refurbish the vehicle in a matter of days. In 1994, control of the program shifted to NASA and the program became known as DC-XA.<sup>22,23,24</sup> The DC-XA design included substantial use of composites, including a composite liquid hydrogen tank. The full-scale Delta Clipper, however, was never funded, as NASA opted to pursue other designs.

The next major SSTO program was the X-33 / Venture Star program. This joint venture between NASA and Lockheed Martin consisted of the sub-scale X-33 demonstrator, to be followed by a heavy-lift SSTO vehicle called Venture Star, capable of placing 50,000 lb<sub>m</sub> [22,680 kg] into LEO.<sup>24</sup> The X-33 was to demonstrate many advanced technologies, including composite cryogenic tanks in complex geometries, highly operable metallic thermal protection tiles, and lightweight linear aerospike rocket engines. Most rockets use a conventional bell nozzle, an axisymmetric converging-diverging structure with the exhaust

contained inside the nozzle contour. Annular aerospike nozzles, in contrast, have a central plug body which is exposed to the ambient environment, over which the exhaust gas is expanded.<sup>11</sup> The linear aerospike uses a plug shape extruded to form a linear expansion surface rather an axisymmetric design. See Figures 2 and 3.

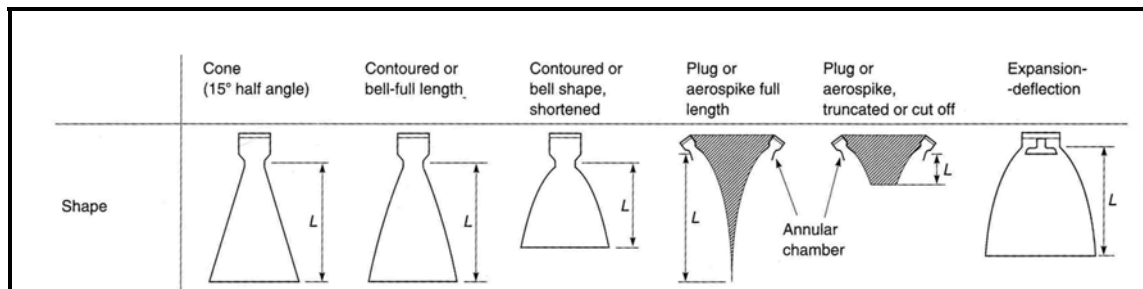


Figure 2. Common Nozzle Geometries (Sutton and Biblarz)<sup>11</sup>

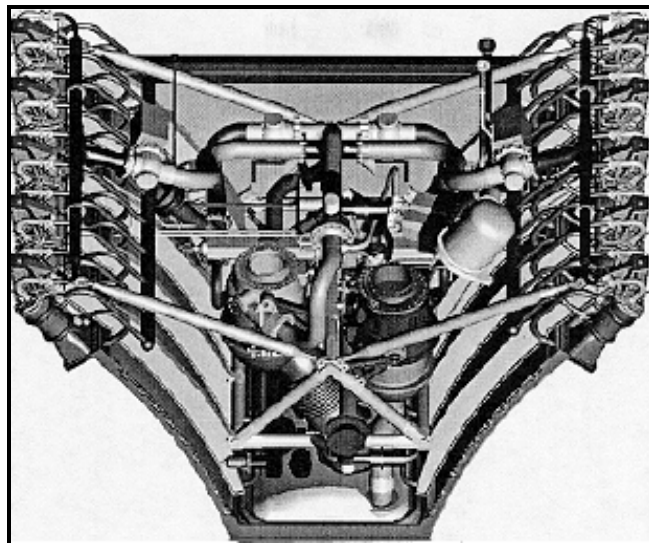


Figure 3. XRS-2200 Linear Aerospike Rocket Engine (Kotake et al.)<sup>25</sup>

The choice of linear aerospike engines was dictated to a large extent by the decision to use a lifting body shape for the vehicle as opposed to a winged body shape similar to the STS Orbiter (see Figure 4). Lifting bodies have been extensively flight tested since the 1970s, and can offer better aerodynamic

performance during hypersonic reentry than winged bodies.<sup>26</sup> The lifting body shape chosen for X-33 / Venture Star favored a linear aerospike engine due to the large linear base area on the aft end of the vehicle. Aerospike engines also offer

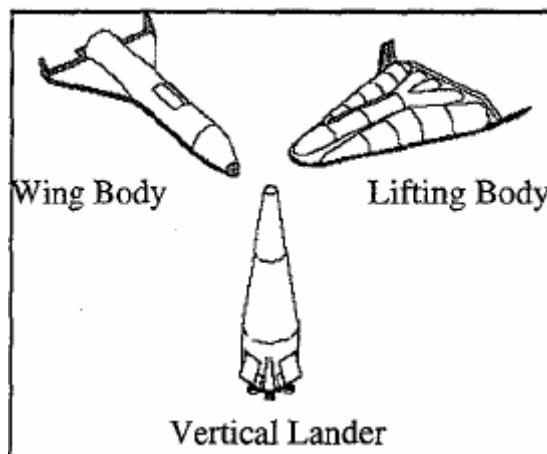


Figure 4. Common Rocket SSTO Configurations (Cook)<sup>27</sup>

the potential for continuous altitude compensation. A rocket nozzle produces maximum thrust when the pressure of the exhaust at the nozzle exit is equal to the pressure of the ambient environment.<sup>11</sup> Most conventional bell nozzles cannot change expansion ratio during flight, so for first stage booster applications, a compromise must be made to select a moderate expansion ratio that is not optimized for sea-level performance or for high-altitude performance, but provides adequate performance throughout the trajectory. Since the aerospike nozzle has no outer wall, the exhaust always expands to ambient conditions and will not suffer performance losses due to atmospheric pressure to the extent that a bell nozzle will.

A drawback of linear aerospike designs is low thrust-to-weight. The XRS-2200 linear aerospike engine developed for the X-33 achieved a vacuum thrust-to-weight of only 35:1, despite using many components from the J-2S, an engine

with a bell nozzle and vacuum thrust to weight of nearly 70:1.<sup>28,29</sup> Had the program proceeded to build the Venture Star vehicle, the linear aerospike engines for the full scale vehicle were required to provide a vacuum thrust to weight of better than 70:1, which was to have been achieved through advancements in materials.

Ultimately, failure of one of the composite hydrogen tanks led to substitution of aluminum tanks in the X-33.<sup>24</sup> Aluminum tanks, weighing considerably more than the composite tanks, lowered the propellant mass fraction. As the X-33 design moved further away from the technologies that would be required to demonstrate Venture Star, NASA support waned and funding was terminated in 2001. The program was subsequently reviewed by the U.S. Air Force, but was not considered applicable enough to military spaceplanes for the service to take over the program.

#### Current Interest in SSTO Launch Vehicles

At the time of this writing, NASA is pursuing the Vision for Space Exploration, with the stated goals of resumption of manned lunar expeditions by 2020, with eventual “human space exploration to Mars and other destinations” at an unspecified future date.<sup>30</sup> The Exploration Systems Architecture Study has recommended expendable, multi-stage heavy lift vehicles derived from STS components to support this Vision.<sup>31</sup> SSTO launch vehicles are not part of the architecture, which is not surprising given the very large payloads that must be delivered (>100 tons to LEO) and the infrequent nature of such missions.

The U.S. Air Force is pursuing Affordable REusable Spacelift (ARES), a multi-stage system with a reusable first stage and expendable upper stage(s).<sup>32</sup> This configuration is anticipated to be the optimum solution for an expected flight



rate of 10-15 flights per year, which is not high enough to justify a fully reusable launch vehicle, but high enough for ARES to be more cost effective than fully expendable vehicles. The mission of the vehicle will be to carry 10,000 to 15,000 lb<sub>m</sub> to LEO, consisting of “tactical space assets and conventional satellites.”

Clearly, there is relatively little interest from U.S. government agencies in pursuing SSTO launch vehicles in the near term. It has been noted, however, that the development of an SSTO launch vehicle remains a long-term objective of many in government and industry.<sup>33</sup> SSTO launch vehicles were part of the roadmap for the NASA initiatives that bridged the gap between X-33 and the current shift in focus toward large expendables. These initiatives were variously known as Space Launch Initiative, Next Generation Launch Technology, and the Integrated Space Transportation Plan, and included SSTO launch vehicles as objectives for the future.<sup>34</sup>

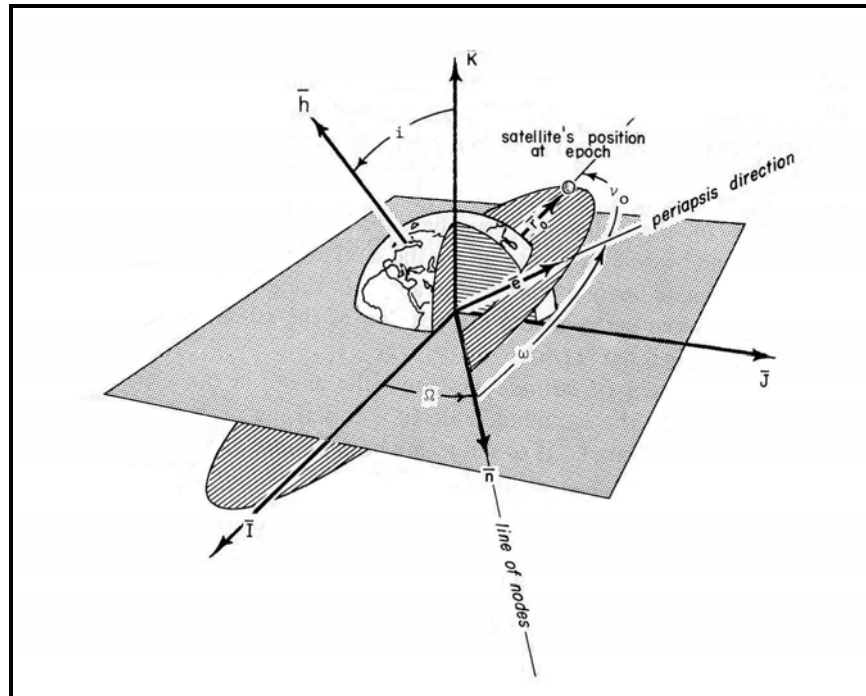
Though SSTO launch vehicle studies are not currently receiving high levels of funding, based on the history of the launch vehicle industry, it is likely that the time will come when such concepts are again in favor. To date, technology has not advanced to point where SSTO flight can be demonstrated. Advocates of various advanced technologies often claim that maturation of their technology will enable SSTO. The objective of this study is to examine the effects that future technology advancements could have towards the goal of enabling SSTO. This objective is accomplished through the development of a weights and sizing model for SSTO vehicles and determination of the effects on SSTO vehicle size of variations in several key performance parameters.

## Chapter 2

### REVIEW OF EXISTING PARAMETRIC ROCKET SSTD SIZING STUDIES

The NASA Langley Research Center has conducted many conceptual studies on reusable launch vehicle architectures, including rocket-based SSTD vehicles. The Advanced Manned Launch System (AMLS) study was conducted in the late 1980s and early 1990s with the aim of finding a less costly alternative to the STS. Freeman et al. have presented details of point design TSTD and SSTD rocket and air-breathing vehicles that would have fulfilled AMLS missions.<sup>21</sup> These missions included delivery of a 30,000 lb<sub>m</sub> [13600 kg] to a 28.5 deg inclination orbit to the then-planned Space Station Freedom and 10,000 lb<sub>m</sub> [4536 kg] payload to a 90 deg inclination orbit. Inclination is defined as the angle between the vector running from the center of the Earth through the North Pole, and the angular momentum vector of the satellite.<sup>35</sup> Inclination is represented by angle  $i$  in Figure 5. Launches into lower inclination orbits can proceed along an easterly launch azimuth that allows the vehicle to take advantage of the Earth's rotation as it gains orbital velocity. Higher inclination orbits require launch along less advantageous azimuths that will increase the delta-V required from the launch vehicle, and will reduce the payload that can be lifted by a given vehicle design. Both missions were to a 50 nm by 100 nm [93 km by 185 km] initial orbit; any additional maneuvering (as would have been required for space station rendezvous) was to be provided by the Orbital Maneuvering System (OMS) propulsion rather than the main propulsion.

The study considered vehicle designs for two different technology epochs: A near-term set of technologies, which at the time were expected be available in

Figure 5. Orbital Elements (Bate, Mueller and White)<sup>35</sup>

the early 1990s, and a considerably more advanced set of technologies with no expected readiness date, but anticipated to be developed under the NASP program. It is interesting to note expectations of maturity for the “near term” technologies by the 1990s have proven to be optimistic. These include replacement of aluminum structure with composite structure, replacement of hydraulics actuators with electro-mechanical actuators, replacement of hypergolic OMS propellants with more benign propellants, and the development of a lighter variant of the Space Shuttle Main Engine (SSME). Though all of these technologies have been studied, to date they have not seen widespread use in operational vehicles.

The chosen vehicle configuration was a winged body, vertical takeoff horizontal landing (VTHL) vehicle with an outer mold line similar to the STS Orbiter. Detailed weight estimates for the study described by Freeman were developed with the code CONSIZ, and optimum ascent trajectories were modeled with the Program to Optimize Simulated Trajectories (POST). Results of this sizing indicated that “near term” technology set could not produce a feasible SSTD design, at least not with the crew escape and engine-out operations requirements that existed for the AMLS missions. The more advanced technology set, which included significant advances in structural materials, as well as a 100% improvement in the vacuum thrust to weight of an SSME-derived engine (existing SSME thrust to weight is 69:1)<sup>36</sup> was able to enable an SSTD vehicle, though the authors cautioned that the cost of developing the advanced technologies may make this option unattractive when compared to a TSTD concept, which was feasible under the “near term” technology set. The study also demonstrated the high growth factor inherent in SSTD designs by reducing the inert weight margin to 10% from the standard 15% and removing crew escape and engine-out provisions from the vehicle and noting the high sensitivity to these perturbations.

A subsequent study by Stanley et al., also from Langley Research Center, examined the effects of varying the specific impulse and thrust to weight ratio of the main engine of a rocket SSTD launch vehicle in order to determine what level of propulsion technology would be required to enable this class of vehicle.<sup>37</sup> The study again supported the AMLS program, though this time the mission was to resupply the notional Space Station Freedom with 20,000 lb<sub>m</sub> [9072 kg] of payload and two crewmembers. The vehicle would also have sufficient cross-range capability during reentry to allow an abort “once-around” trajectory from a polar orbit, and the chosen shape was again a winged body. The structural technology

was described as “evolutionary” and included significant use of graphite and carbon-carbon composites, though the tanks and thrust structure would remain aluminum-lithium. The baseline engine concept is an SSME-derived LOX/LH<sub>2</sub> engine with a vacuum specific impulse of 453.5 sec and a vacuum thrust to weight ratio of 100:1. Baseline vehicle thrust to weight ratio at liftoff was set at 1.2.

Stanley et al. identify the three most important propulsion parameters affecting the feasibility of a near-term SSTO as specific impulse, thrust to weight ratio of the main engines, and bulk density of the propellants. The study addressed the effects of specific impulse and thrust to weight ratio, but did not thoroughly consider bulk density effects. Curves depicting the variations in vehicle GLOW and dry weight as specific impulse and thrust to weight were produced using CONSIZ for vehicle weight estimation and POST for trajectory simulation, as was the case in the previous study. The dry weight curves from this study are presented in Figure 6.

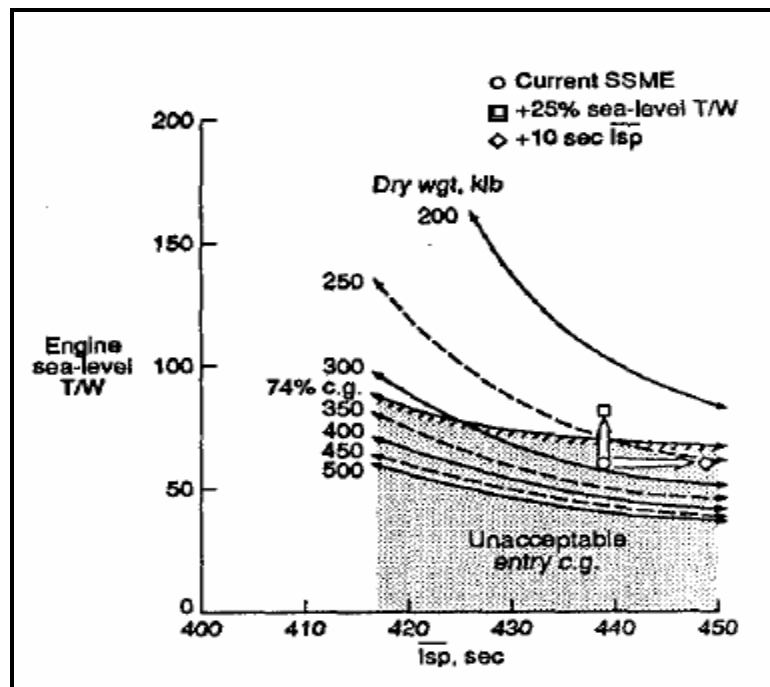


Figure 6. AMLS Dry Weight Sensitivity to Isp and T/W (Stanley et al.)

Since the engine is located at the aft end of the vehicle, if engine thrust to weight ratio is low enough, the vehicle center of gravity will move so far aft that the vehicle can not be trimmed for flight. Stanley et al. used these curves to evaluate a variety of different LOX/LH<sub>2</sub> engines with various chamber pressures and expansion ratios (and therefore various specific impulse and thrust to weight values) without the need to perform detailed sizing analysis. The curves were also used to evaluate the relative merits of a 10 second specific impulse improvement to the existing SSME versus a 25% dry weight reduction to that engine. From the data produced, it was apparent that it was advantageous to invest in thrust to weight improvements for this application, even if specific impulse were reduced as a consequence.

Stanley et al. also investigated the effecting of varying the engine O:F ratio. A higher O:F than the point of departure of 6:1 will use more of the dense liquid oxygen and raise the bulk density of the propellants, but the leaner mixture reduces the specific impulse. Using the sizing methodology of this study, Stanley et al. determined that an O:F ratio of 6.5:1 yielded the lowest dry weight. The study also considered several other engine configuration issues, including changing the mixture ratio during flight, using a two-position nozzle for greater expansion ratio at altitude, optimum lift-off vehicle thrust-to-weight, and engine gimbaling considerations.

The conceptual design studies conducted at NASA Langley Research Center for the Advanced Manned Launch System thoroughly explored the tradespace for LOX/LH<sub>2</sub> SSTO launch vehicles that used advanced structures. This project will build on that work, adding a degree of freedom for propellant bulk density. This will allow propellants with arbitrary properties to be evaluated for SSTO applications.

## Chapter 3

### ADVANCED PROPELLANTS

Clearly, propellant combinations with high specific impulse and bulk density are highly desirable. As shown in Table 1, however, propellants with high values for one property often have low values for the other. Propellant chemists have worked in many different research areas in attempts to create propellants with density and specific impulse exceeding that of current propellants.

#### Propellant Performance

The theoretical specific impulse that can be attained by a rocket is given by the equation

$$I_{sp} = \frac{1}{g_0} \sqrt{\frac{2\gamma}{\gamma-1} \frac{R'T_0}{M} \left[ 1 - \left( \frac{p_2}{p_1} \right)^{(\gamma-1)/\gamma} \right]}$$

where  $T_0$  is the chamber temperature and  $M$  is the molecular weight of the exhaust products.<sup>11</sup> Specific impulse can be increased by raising chamber temperature and decreasing the molecular weight of the exhaust species. Using this equation, specific impulse has been plotted as function of exhaust species molecular weight for several different chamber temperatures in Figure 7.

There are practical limitations to the chamber temperature that can be achieved. High-performance engines use film cooling inside the chamber against the wall and coolant passages outside the chamber wall to allow chamber

temperatures as high as 7000°R (3889 K).<sup>11</sup> Significantly higher chamber temperatures would not be possible with present cooling techniques.

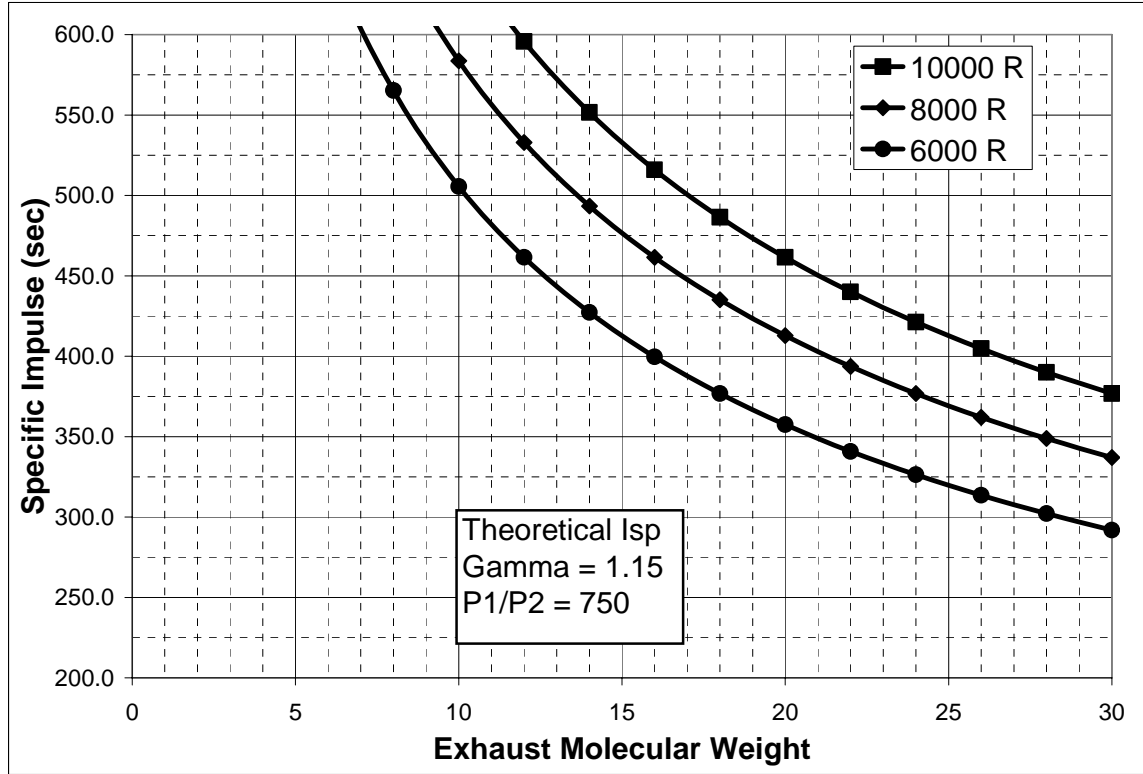


Figure 7. Theoretical  $I_{sp}$  as a function of  $M$  and  $T_0$

#### Adiabatic Flame Temperature

The maximum temperature that a combustion process can reach is the temperature that would occur if the reaction takes place in a closed, adiabatic system.<sup>38</sup> This temperature is known as the adiabatic flame temperature. Under these assumptions, all of the energy released by the reaction will be used to bring the products to the flame temperature. This flame temperature can be found by solving the equation

$$\sum_{i=1}^N v_i'' \Delta H_{f, M_i}^0 + \sum_{i=1}^N v_i'' (H_{M_i, T_f} - H_{M_i, 298}) = \sum_{i=1}^N v_i' \Delta H_{f, M_i}^0 + \sum_{i=1}^N v_i' (H_{M_i, T_i} - H_{M_i, 298})$$



where  $v'$  represents the stoichiometric coefficient of reactant species,  $v''$  represents stoichiometric coefficients of product species,  $\Delta H_f^0$  represents the heat of formation at standard conditions, and  $H$  represents enthalpy. Solving this equation is difficult due to several complications. First, the equation must be solved by iteration as there are many enthalpy terms that depend on temperature which can not be simplified. In addition, the stoichiometric coefficients of the reaction can change as temperature changes. At high temperatures, some of the molecules may dissociate into a variety of different compounds that would not exist in appreciable amounts at lower temperatures. The dissociation reactions are dictated by equilibrium constants that depend on the free energy of formation of the various species.

Before computers were widely available, calculation of the adiabatic flame temperature was very tedious and time-consuming.<sup>16</sup> Today, the computer program CEA allows for rapid, accurate solution of these types of problems.<sup>15</sup> Using CEA, the molecular weight of the combustion products for LOX/LH<sub>2</sub> at an O:F of 6 and chamber pressure of 3750 psia [25.9 MPa] is found to be on the order of 13.7, as the products are primarily H<sub>2</sub> and H<sub>2</sub>O. The program also provides a chamber temperature of 6516°R [3620 K] for these conditions, and the theoretical specific impulse of this combination is in excess of 460 sec. for these conditions as shown on Figure 7.

### Properties of the ideal propellant

Clark<sup>16</sup> has identified these properties for the exhaust species of the ideal propellant: High heat of formation, low molecular weight, and low  $C_p$ . High heat of formation will ensure plenty of energy is provided to raise chamber temperature, low molecular weight will allow efficient transfer of the thermal energy into kinetic energy in the nozzle, and low  $C_p$  will allow for higher chamber temperatures for a given energy release from the reaction. Unfortunately, no one chemical species has all of these properties. The species that make the best compromise among these qualities include helium and hydrogen. Helium, of course, as a noble gas, is not suitable for chemical rockets. With an energetic oxidizer, such as oxygen or fluorine, specific impulse with hydrogen fuel can be very high, as shown in Table 1.

### High Energy Density Materials

Of course, the need for low molecular weight exhaust products is contrary to the vehicle-level need for dense propellants for high propellant mass fraction, as dense reactants will tend to create dense exhaust products. Over the years there have been many different attempts to create propellants with both high energy content and high density. Clark<sup>16</sup> provides details on many of these efforts from the 1950s and 1960s. The following examples are notable attempts, past and present, to create high energy density propellants.

### Lithium-Fluorine-Hydrogen Tripropellant

Clark briefly discusses this propellant combination, and it has the distinction of being among the highest specific impulse of any chemical rocket ever tested. The testing involved liquid fluorine (which is highly toxic, as

mentioned above), liquid lithium (which is liquid only at high temperature, reacts violently with air, and is highly corrosive), and gaseous hydrogen.<sup>16</sup> Thus there is little hope of practical application of this technology due to operations and safety considerations, but the case is still instructive in showing what can be accomplished with chemical propulsion. Two different injection schemes were used: The first used the  $\text{LF}_2$  and a very small portion of the  $\text{GH}_2$  as a gas generator to produce hot  $\text{GF}_2$  which was reacted with the lithium, and the rest of the hydrogen was injected downstream. The other method was to mix the  $\text{LF}_2$  with all of the  $\text{GH}_2$ , producing hot gaseous  $\text{HF}$  to react with the lithium.<sup>39</sup> The chamber was cooled with an external water source.

A vacuum specific impulse of 506 sec. was measured for an engine with chamber pressure of 750 psia [5.2 MPa] and a nozzle expansion ratio of 60:1. Arbit et al. estimate that for an engine regeneratively cooled with hydrogen, operating at 1000 psia [6.9 MPa] and a nozzle expansion ratio of 100:1, the vacuum specific impulse could be 523 sec.<sup>39</sup> The addition of a lightweight, reactive metal to an already energetic propellant combination was able to produce better specific impulse and higher bulk density, but again, the hazardous nature of these propellants precludes their use in a space launch vehicle.

### Strained Ring Hydrocarbons

Many hydrocarbon fuels possess relatively high densities. In order to increase the specific impulse possible with these fuels, isomers with superior heats of formation can be created. The high heat of formation is due to the unique arrangement of the atoms and the higher bond energies that result.<sup>40,41</sup> However, this higher energy is usually accompanied by a lower activation energy required for decomposition, which means a less stable fuel. Bai et al. have described small

scale hot-fire rocket engine testing using several fuels of this type.<sup>42</sup> Fuels tested included quadricyclane ( $C_7H_8$ ) and bicyclopropylidene ( $C_6H_8$ ), or BCP. Table 2 lists the bulk densities and specific impulses that could be achieved for these propellants, using the same assumptions listed for the data in Table 1.

Table 2. Properties of Representative Strained Ring Hydrocarbon Propellants

Oxidizer	Fuel	Oxidizer Density (lb/ft <sup>3</sup> )	Fuel Density (lb/ft <sup>3</sup> )	O:F Ratio	Bulk Density (lb/ft <sup>3</sup> )	Vacuum I <sub>sp</sub> (sec)
LOX	$C_7H_8$	71.2	61.1	2.4	67.9	356.4
LOX	$C_6H_8$	71.2	52.8	2.4	64.6	359.7

Either of these fuels will compare favorably to RP-1 or methane if performance is all that is considered. However, before a novel fuel can transition into operational use, questions of toxicity, material compatibility, stability, and producibility must be considered. At the time of this writing, neither of these fuels had completed all of those steps on the path to operational use.<sup>43</sup>

### Polynitrogen Compounds

For many years, nitrogen compounds besides  $N_2$  and  $N_3^-$  have been the subject of theoretical studies, as the decomposition of polynitrogen compounds such as  $N_4$ ,  $N_6$ , or  $N_8$  to several molecules of  $N_2$  could form the basis for high energy-density materials.<sup>44</sup> Successful synthesis of a compound containing  $N_5^+$  was reported by Christe et al. in 1999 and has led to increased efforts to synthesize a polynitrogen compound that is useful as a propellant.<sup>45</sup> Dixon et al. have calculated the theoretical heats of formation for several polynitrogen ions and have

investigated the stability of several polynitrogen salts.<sup>46</sup> This paper as well as another by Fau et al.<sup>47</sup> estimate the heat of formation for the  $N_5^+$  ion to be near 350 kcal / mol. The radical  $N_5^+$  will readily join with an electron so it alone cannot form a practical propellant. If it could, however, CEA calculations indicate that the theoretical specific impulse would be in excess of 580 sec. Even if the ion were a feasible propellant, the high molecular weight of the exhaust products (which will consist of a mixture of  $N_2$  and  $N$ ) indicate that such high values for specific impulse would only be possible with chamber temperatures above 16500°R [9200 K], which are far higher than any known materials could withstand. Research is directed at finding an appropriate pair of ions to form a polynitrogen salt or possibly an ionic liquid.<sup>48</sup> Dixon et al. have shown that theory predicts both  $N_5^+N_5^-$  and  $N_5^+N_3^-$  will not be stable in the solid form, as the activation energies are quite low.

Density estimates for polynitrogen compounds have ranged from 93.6 lb/ft<sup>3</sup> [1.50 g/cm<sup>3</sup>] predicted by Dixon et al. for  $N_5^+N_5^-$  to over 137 lb/ft<sup>3</sup> [2.20 g/cm<sup>3</sup>] for polynitrogen species such as  $N_{10}$  or  $N_{12}$  predicted by Haskins et al.<sup>49</sup> These values are higher than any liquid propellants in common use and are similar to densities achieved with solid rocket motor propellants.<sup>11</sup>

If a stable combination can be found, polynitrogen propellants could offer a combination of high specific impulse and good bulk density from a monopropellant (no oxidizer is required). As described above, any polynitrogen propellant would also have to be relatively easy to manufacture, compatible with aerospace materials, preferably non-toxic, and preferably a liquid at ambient temperatures for ease of storage if it were to be adopted for operational use.

## Chapter 4

### PARAMETRIC SSTO VEHICLE SIZING METHODOLOGY

The purpose of this study is to determine the sensitivity of SSTO rocket launch vehicle size to specific impulse, propellant bulk density, and main propulsion thrust to weight ratio so that the applicability of advanced propellants for the SSTO mission can be assessed. Before that work could proceed, appropriate limits on these parameters were chosen, and launch vehicle mission and operations concept were chosen. An existing U.S. Air Force vehicle weight estimation program was then adapted to reflect those assumptions. This code was then used to produce weight estimates for every combination of design variables in the design of experiments, and trajectories of each of these SSTO designs were simulated to confirm the vehicle was correctly sized.

#### Study Parameters and Design of Experiments

As described above, the three main parameters of interest for this study are specific impulse, propellant bulk density, and main propulsion thrust-to-weight ratio. SSTO vehicles, as described above, can be very sensitive to the assumptions made about structural technology. The heavy insulation required on the propellant tanks for “deep” cryogenic fluids such as liquid hydrogen can also have a significant impact on the propellant mass fraction as described above. In order to capture these effects, structures technology and tank insulation weight were also selected as parameters for the study.

### Specific Impulse

Vacuum specific impulse was allowed to vary from 350 sec. to 550 sec. in order to capture the range of values of potential propellants, from LOX with hydrocarbon fuels at the lower end to polynitrogen compounds or tripropellants with liquid metals at the higher end of the range.

### Bulk Density

Bulk density was allowed to vary from 20 lb<sub>m</sub>/ft<sup>3</sup> [0.32 g/cm<sup>3</sup>] to 120 lb<sub>m</sub>/ft<sup>3</sup> [1.92 g/cm<sup>3</sup>] to represent the range from LOX/LH<sub>2</sub> at the lower end to potential polynitrogen compounds at the higher end of this range. Since this study is intended to be used to estimate vehicle weights for arbitrary propellants, O:F ratio is not known *a priori*. An O:F ratio of 6.0 was assumed for all bulk densities, and the densities shown in Table 3 were chosen for oxidizers and fuels. Sensitivity analysis has shown that vehicle size was not sensitive to the O:F ratio and the individual densities of the oxidizer and fuel, but rather to the bulk density.

Table 3. Oxidizer and Fuel Densities Selected for O:F of 6.0

Oxidizer Density (lb <sub>m</sub> /ft <sup>3</sup> )	Fuel Density (lb <sub>m</sub> /ft <sup>3</sup> )	Mixture Ratio	Bulk Density (lb <sub>m</sub> /ft <sup>3</sup> )
60.0	4.0	6.0	20.0
60.0	18.0	6.0	45.0
120.0	20.0	6.0	70.0
120.0	42.2	6.0	95.0
120.0	120.0	6.0	120.0

### Engine Thrust to Weight Ratio

Engine vacuum thrust to weight ratio was varied from 50:1 to 150:1, representing the range from a 50% weight growth for the SSME to 50% weight reduction on the SSME. Note that the design of turbopumps, a significant component of engine weight, are heavily influenced by the density of the propellant being pumped.<sup>11,50</sup> Propellants with low density will require engines with more powerful turbopumps and, for a given technology level, will generally have a lower thrust to weight than an equivalent engine using denser propellants.

### Structures Technology and Cryogenic Insulation

Each combination of specific impulse, bulk density, and thrust to weight ratio was assessed for four different assumptions concerning structures technology and cryogenic insulation: Near-term structures with minimal insulation, near-term structures with heavy insulation, advanced structures with minimal insulation, and advanced structures with heavy insulation. A full factorial design of experiments was chosen, resulting in 300 different point designs to be assessed.

Vehicles sized with near-term structures technology assumptions used metallic and composite airframe components and metallic propellant tanks. Vehicles sized with advanced structures technology assumptions used more composites and fewer metallic items in the airframe, as well as composite propellant tanks. Further details on these assumptions are provided below.

The assumption of minimal insulation was represented by an areal insulation density of  $0.1 \text{ lb}_m/\text{ft}^2$  [ $0.004 \text{ kg}/\text{m}^2$ ] on both propellant tanks, while heavy insulation typical of deep cryogenic liquids such as liquid hydrogen was represented by an areal density of  $1.0 \text{ lb}_m/\text{ft}^2$  [ $0.04 \text{ kg}/\text{m}^2$ ]. For actual propellants, the insulation thickness may not be the same for oxidizer and fuel (LOX/LH<sub>2</sub> is



one example). However, it would not have been possible to capture the full effects of different insulation thicknesses on the oxidizer and fuel tanks unless O:F ratio was varied as well. As this would have added another parameter that would have increased the number of point designs by an order of magnitude, that level of complexity was not possible in the time available for this study.

### Mission and Operations Concept

Space launch vehicles generally support one of three uses: Lifting commercial satellites for profit, supporting space exploration, through launch of humans or robots in support of science missions (typically sponsored by a government), or supporting the needs of a nation's military by placing satellites into orbit. The mission and operation concept chosen for the SSTO vehicle in this study conform to reasonable expectations for a military launch vehicle, and there are several reasons why that mission was chosen over the other two.

As described above, it is difficult for reusable launch vehicles be an economical alternative to expendable launch vehicles unless the system is used at a high tempo. Projections by Futron Corporation suggest that the worldwide market for small ( $< 5000 \text{ lb}_m$ ) satellites is unlikely to exceed 20 launches per year for the foreseeable future.<sup>51</sup> Furthermore, there is an oversupply of launch vehicle capacity which is only predicted to grow. Some of these vehicles are foreign launch systems that can offer extremely low prices. Although this market segment has been cited in the past as the appropriate market for a reusable commercial launcher to enter, collapse of the LEO telecommunications satellite market has removed the need for additional launch vehicles of this class.<sup>24,52</sup> Larger commercial satellites are also well served by existing launch vehicles. Over the five-year period from July 2000 to June 2005, the most widely used vehicles were

the Ariane 4, Delta II, and Proton launch vehicles, each with 19 missions, and the Ariane 5 with 16 launches.<sup>53</sup> Together, all of these vehicles' manifests add up to less than 15 missions per year, which is unlikely to be enough to enable economical reusable launch vehicles.

As described above, NASA is moving towards a mission best suited to very heavy expendable vehicles. In addition, carrying a crew requires a number of extra subsystems to be installed on the vehicle that would not be required for other missions. Crewed missions are often of long duration compared to other launch vehicle missions. Shuttle missions routinely last up to 14 days.<sup>2</sup> In order to remain in orbit for that long, such a mission will require the vehicle to orbit at an altitude of at least 150 nm [280 km] to avoid early reentry due to atmospheric drag.<sup>35</sup> The vehicle will then require a system like the OMS to perform a deorbit burn at the desired time. A vehicle that merely delivers payload to orbit, in contrast, could enter a "once-around" orbit that will reenter on the first orbit, negating the need for OMS, thus improving propellant mass fraction. The payload could be transferred from the delivery conditions to the target orbit with its own propulsion or a small rocket stage. A crewed vehicle will require environmental and life support systems for the long-duration mission; the military vehicle needs no crew. The long-duration mission will require more propellant for the reaction control systems (RCS) than the once-around mission. Crewed vehicles will need to be "human-rated," and will need additional systems to provide realistic abort scenarios to rescue the crew from a catastrophic systems failure, over as large a portion of the flight regime as is possible. This usually takes the form of escape capsules, multiple engines that can compensate for off-nominal performance from any one engine, and triple-redundant backups of primary systems.<sup>54</sup>

In addition to the weight impacts of the additional systems that must be carried by a crewed vehicle, there are also operations impacts associated with these systems. The Space Propulsion Synergy Team has identified “lessons learned” from STS operations experience in the form of desirable characteristics for future reusable launch vehicles to reduce operations costs and refurbishment timelines.<sup>55</sup> The top recommendation of the team was to reduce the number of toxic fluids on the vehicle. Highly toxic fluids used in the OMS/RCS, Auxiliary Power Units (APUs), environmental/cooling, and other systems seriously impact the operation of the Orbiter, as non-essential personnel must be cleared away from the Orbiter when any of these systems are serviced. A once-around, unmanned vehicle could eliminate the OMS and severely reduce the need for RCS, APUs, and environmental systems. Other high priority objectives that are easier to meet with the once-around vehicle include reducing the number of propulsion systems, reducing the number of fluid connections (due to reduced redundancy requirements), number of components required for successful flight operations, and number of parts. The once-around vehicle can be easier to maintain than the crewed vehicle, making the prospect of achieving a high flight rate more realistic.

#### Military Spacelift Mission

Military satellites currently have a life cycle very similar to other satellites. They are designed for long lifetimes and they are launched on expendable launch vehicles, under a contract that may have been arranged years in advance. In recent years, there has been a great deal of interest in changing this paradigm to one of putting satellites in orbit on very short notice to fulfill the rapidly changing needs of military users on the ground, sea, or in the air. This paradigm is perhaps best described by Rosen et al. in a paper from 1990: “A tactical satellite, or TACSAT,

is a satellite that can be responsively employed to support the specific needs of tactical forces... to provide satellite service during critical periods when the demand is great than the supply.”<sup>56</sup> Today, this type of spacelift mission is referred to as “operationally responsive.”<sup>57,58</sup> Potential flight rates for this system (though they may come in unpredictable surges) could be high enough to justify a reusable system, if such a capability is deemed necessary.

SSTO Vehicle Mission The design reference mission chosen for this study is an operationally response military spacelift mission. The payload of the system will be 10,000 lb<sub>m</sub> [4536 kg] to a 56 nm by 118 nm [100 km by 219 km] polar orbit. This mission is very similar to one of the reference missions for the AMLS and is consistent with other recent military launch vehicle studies.<sup>22,59</sup> Although many envision tactical satellites to be small in size, Rosen et al. emphasize that “TACSATS are not necessarily ‘smallsats’ or ‘cheapsats;’ rather they are sized to perform the needed service.” The payload of 10,000 lb<sub>m</sub> is therefore appropriate when one considers that some of the satellites will require apogee kick stages to reach higher energy orbits and some missions may deploy more than one satellite at a time.

#### Weight Estimation Techniques

Weight estimation for the various SSTO point designs was accomplished using a spreadsheet-based tool derived from the Air Force Weights Analysis Tool (AFWAT), which can estimate the weights of various subsystems for conceptual air-breathing or rocket based SSTO and TSTO vehicles. AFWAT is based on a compilation of numerous Air Force and NASA weight estimating programs, including the CONSIZ code used in the AMLS study. AFWAT is not designed to accept arbitrary propellant properties, and is not designed to size a vehicle to meet

a design reference mission. A spreadsheet-based vehicle sizing tool was developed that is capable of sizing a vehicle with arbitrary propellants, using many of the AFWAT mass estimating relationships to provide the weights of various subsystems. The assumptions used to estimate masses for each of those subsystems is described below.

### Aerodynamic Surfaces

This section includes the wing, elevons, vertical tail, and the body flap, a control surface on winged body vehicles that protects the rocket engines from reentry heating and can be deflected to aid in trimming the vehicle during gliding flight. As the Orbiter has been a highly successful winged body design, wing parameters were set to match that vehicle. Key among these is the wing loading, the ratio of the weight of a vehicle during flight to the wing surface area. Vehicle aerodynamic stall speed is directly proportional to the square root of wing loading.

$$V_s = \sqrt{\frac{2W}{\rho S C_{L_{\max}}}}$$

In the equation above,  $W$  is vehicle weight,  $\rho$  is the density of the atmosphere,  $S$  is the wing reference area, and  $C_{L_{\max}}$  is the maximum lift coefficient of the vehicle.<sup>60</sup> Wing sizing for the Orbiter is a compromise between the desire for low stall speed for ease of landing (larger wing for lower stall speed) and minimizing the area of the TPS tiles (smaller wing, fewer tiles). Wing loading for this study is based on that chosen for the Orbiter.<sup>61</sup> Other parameters that affect the estimation of wing weight include the leading edge sweep angles, the aspect ratio of the wing, and the thickness to chord ratio of the wing. All of these variables were set to be close to Orbiter values. Material assumptions for the “near-term” structures technology

included metallic/composite honeycomb structure for the wings and a graphite epoxy tail. For advanced structures, the wing structure is an all-composite honeycomb structure.

### Airframe Structure

This category included nose cone structure, intertank structure, thrust structure, and the aft body section. Nose cone and aft body weight are a function of vehicle body diameter and the maximum dynamic pressure the vehicle will encounter, assumed to be  $700 \text{ lb}_f/\text{ft}^2$  [33.5 kPa], slightly higher than that experienced by the STS;<sup>2</sup> intertank weight is a function of vehicle body diameter and maximum thrust of the vehicle, and thrust structure weight is function of maximum thrust of the vehicle.

### Thermal Protection System

TPS included insulation internal to the fuselage as well as material applied to the outside of the vehicle. Internal TPS weight was a function of the length of the propellant tanks; while external TPS considered the geometry of the wings, the wetted area of the entire vehicle, and the areas of the vertical tail, body flap, and landing gear bays. Current research on advanced metallic TPS suggests that while metallic TPS will be more operable than the Orbiter's ceramic tiles and thermal blankets, it will not be significantly lighter.<sup>62</sup> TPS weight was therefore not considered a function of structural technology level.

### Landing Gear

This category included weight of the running gear and control mechanisms, and is a function of the landing weight of the vehicle. Landing weight is the sum

of all vehicle dry weights, the payload (vehicles may have to land with the payload for aborted missions or missions to retrieve satellites), and residual propellants.

### Rocket Engine

Rocket engine weight was calculated by dividing the vacuum thrust required for the vehicle by the assumed thrust to weight ratio for each point design. The sea level thrust requirement was 1.2 times the vehicles weight at lift off, based on optimum levels established by Stanley et al. No engine-out capability was assumed for this vehicle, as it is unmanned. Sea level thrust is related to vacuum thrust by the equation

$$F_{SL} = F_{vac} - A_{exit} \cdot P_{SL}$$

where  $A_{exit}$  is the nozzle exit area and  $P_{SL}$  is the ambient pressure at sea level. All engines were assumed to operate at a chamber pressure of 3750 psia [25.9 MPa] with a nozzle expansion ratio of 70:1. This is a higher chamber pressure than the SSME, but similar to high-performance Russian engines such as the RD-180.<sup>63</sup> At this pressure, a 70:1 nozzle is near the maximum that can be used without flow separation at sea level, which has serious performance consequences.<sup>11</sup> Engine thrust is given by the equation

$$F_{vac} = C_{Fvac} \cdot A_{throat} \cdot P_{chamber}$$

where  $C_{Fvac}$  is the thrust coefficient, a parameter that depends on the specific heat ratio  $\gamma$  of the exhaust and the nozzle expansion ratio.<sup>11</sup> A thrust coefficient of 1.9 was assumed for all point designs, which corresponds to a specific heat ratio of approximately 1.15, and should give reasonable results for arbitrary liquid propellants.

### Orientation Control Systems

This category included aerodynamic control surface actuators, as well as the RCS. No OMS weight was included as the once around vehicle does not require this system. Aerodynamic control surface actuator weights are functions of the control surface areas. RCS weights assumed that RCS propellants had properties similar to the hypergolic (and toxic) nitrogen tetroxide oxidizer and monomethylhydrazine fuels used by the Orbiter. Although using the same propellants for RCS as for the main engines has been proposed as a way to eliminate the hypergolic propellants, this has yet to be flown on operational vehicles,<sup>64,65</sup> and may not be feasible for all propellants, so this approach was not considered for this study. RCS delta-V budget was based on AFWAT default values. RCS weight categories included thrusters, titanium propellant tanks, feed lines, pumps, and propellant.

### Power Supplies

Power supply weight estimates assume that the vehicle has 3 APUs sized to provide main power to the vehicle and four 5kW fuel cells for auxiliary power. Main power requirement was a function of vehicle landing weight. The weight estimate included the APUs, tanks and plumbing for APU propellants, and the fuel cells.

### Main Propellant Tanks

This category includes the tanks for the main propulsion oxidizer and fuel, including anti-slosh baffles, structural members, boost pumps, and pressurization systems. Tank weights were a function of propellant density, propellant required, structural technology assumptions, and cryogenic insulation assumptions. Tank structure had an areal density of  $2.5 \text{ lb}_m/\text{ft}^2$  for near term aluminum tanks, and this



density was reduced by 50% for the advanced structures to represent potential weight savings of composite propellant tanks.<sup>66</sup> Similarly, baffles and structural members were reduced in weight by 50% for advanced technology vehicles. Pressurization system weight was a function of propellant volume. To ensure that vehicles with radically different bulk densities remained geometrically similar for ease of comparison, all vehicles had tank stacks sized to be 5.5 times longer than their diameter.

### Avionics

Several avionics subsystems were fixed weights that were not sensitive to vehicle design. Subsystems that were affected by vehicle design include actuator control systems, which was a function of aerodynamic control surface areas and vehicle vacuum thrust, range safety ordnance which was a function of vehicle length, thermal management systems, which were a function of vehicle length, and cabling, which was a function of vehicle length and diameter.

### Vehicle Health Management

AFWAT uses a fixed mass of 620 lb<sub>m</sub> [281 kg] for vehicle health management. No reusable launch vehicle has had sufficient use of vehicle health management to provide a basis for a better mass estimating relationship. Since such a system would likely include many thousands of small sensors,<sup>67</sup> it is reasonable to expect this weight should be a function of vehicle dimensions.

### Hydraulic Systems

Although replacing hydraulic systems with electromechanical actuators could yield operability benefits by removing a fluid from the vehicle, this technology has not yet been demonstrated on a large scale. Hydraulic systems

weight consists primarily of the distribution lines for hydraulic fluid, and was a function of the length and landing weight of the vehicle.

### Growth Margin

Vehicles in early stages of the design process often carry growth margin as part of the vehicle weight, as weight estimates of various vehicle subsystems inevitably grow as the project matures. Growth margin was assumed to be 15% of the weight of all the vehicle subsystems.

### Drag Chute

As the vehicle will have a wing loading similar to the Orbiter, landing speeds will also be similar, on the order of 200 kts [100 m/s].<sup>68</sup> With such a high landing speed, and very few control surfaces on the wings to aid in slowing the vehicle, a drag chute will be required to stop in a reasonable distance. This weight is fixed in AFWAT as 245 lb<sub>m</sub> [111 kg].

### Payload Bay

The payload bay mass estimating relationship from AFWAT was used, assuming that the payload bay volume required was 1000 ft<sup>3</sup> [28.3 m<sup>3</sup>]. This provides a payload bay weight of 2648 lb<sub>m</sub> [1201 kg] for all designs.

### Propellant Residuals & Reserves

A rocket vehicle's tanks hold more propellant than will actually be burned in the engines. Like any flight vehicle, a performance reserve is included in the event that more propellant is required to perform the mission than expected. A rocket's turbopumps can rotate at very high speeds, producing in excess of 100,000 hp [75 MW].<sup>11</sup> If a rocket's turbopumps run out of propellant, the sudden change can cause the blades to exit their housing in a catastrophic manner.

Additional propellant is required to prevent this from occurring. Also, with any plumbing system, there will always some amount of trapped propellant that can not be expelled. For these reasons, all vehicles are assumed to carry an additional 2% of their propellant load than is not useful towards achieving mission delta-V.

### Vehicle Sizing Techniques

Since many of the subsystem weights depend on the vehicle dry or gross weights, a change in one subsystem can affect many other seeming unrelated systems. These effects can be modeled using the circular reference feature of Microsoft Excel. While the mass estimating relationships described above can provide the subsystem weights for a given vehicle design, they can not properly size a vehicle to perform the desired mission without user intervention.

Usable propellant mass was chosen as the independent variable for the purpose of sizing the vehicles. As the ratio of a cylinder's volume to surface area is one half of the radius, propellant tanks become more efficient as they become larger (pump-fed rocket engines store propellant at low enough pressures that tank thickness is typically determined by manufacturing considerations rather than by pressure vessel burst considerations). As some subsystem weights are fixed, in general the propellant mass fraction of a rocket stage increases as its size increases. For a given specific impulse, the delta-V delivered by the stage will also therefore increase as the stage grows in size.

For each point design, usable propellant mass was manually varied until the vehicle delta-V matched the required mission delta-V. For point designs with low specific impulse, bulk density, or thrust-to-weight ratio, there may be no feasible SSTO vehicle that can achieve the required delta-V. Vehicles requiring more than  $1.0 \times 10^7 \text{ lb}_m$  [ $4.5 \times 10^6 \text{ kg}$ ] of usable propellant were considered to be “unclosed”

designs for which no practical solution existed. Note that a vehicle this large is considered feasible only in the sense that a vehicle of that scale could be constructed; from an economic and engineering standpoint such a design would be highly undesirable. A sample mass estimate for a closed vehicle is provided in Appendix A.

To confirm the sizing, an ascent trajectory for each point design was simulated using the Program to Optimize Simulated Trajectories (POST), the same code used by NASA Langley Research Center for the AMLS studies. The POST Utilization Manual described POST as a “generalized point mass, discrete parameter target and optimization program.”<sup>69</sup> POST was developed in 1970 to simulate Space Shuttle ascent and reentry trajectories. POST is a 3-degree of freedom (3-DOF) simulation,<sup>70</sup> meaning that while it ensures that the vector sum of the forces acting on the vehicle equals the time rate of change of linear momentum to accurately simulate translational motion, it does not ensure that the sum of the moments on the vehicle are equal to the time rate of change of angular momentum, so the onus is on the user to check that any rotational motion performed by the vehicle is within reason.

Forces considered by POST include Earth’s gravitational field (accounting for the oblate shape of the Earth and uneven mass distribution), thrust from the rocket engine, including losses due to atmospheric backpressure, and aerodynamic lift and drag forces. Lift and drag forces are calculated using an input table that provides lift and drag coefficients as a function of flight condition. STS Orbiter aerodynamics tables provided in the POST manual were used as the SSTD vehicle geometry is intended to be similar to that of the Orbiter. A wing reference area calculated by the spreadsheet sizing tool is used in conjunction with the STS Orbiter aerodynamics.

A POST optimization problem consists of an objective function (an output variable to be maximized or minimized), independent variables which the program can vary, and constraints. For the SSTO ascent trajectories, the independent variables were a series of pitch rates that the vehicle will use as steering commands, and the weight of the payload. The objective function is to maximize vehicle weight. This will allow the program to find the trajectory that is most efficient in order to lift the greatest payload possible. Constraints include the altitude, velocity, and flight path angle at burnout to achieve the desired orbit, as well as the maximum dynamic pressure allowed during flight, and limits on the maximum and minimum angle of attack. These angle of attack limits are required since, as a 3-DOF simulation, POST will not verify that the steering commands keep the vehicle at a realistic attitude.

POST also requires input data describing the initial position and velocity of the vehicle, as well as the usable propellant mass and gross mass of the vehicle. Vacuum thrust, specific impulse, and nozzle exit area are also required inputs. An acceleration limit of 15g was applied to these trajectories. As the vehicle burns propellant, if thrust is held constant, acceleration will increase. This limit will throttle the engines to lower power settings to maintain 15g acceleration if it is reached. Some launch vehicles maintain significantly lower acceleration limits, for comfort and safety on manned missions and to protect delicate payloads on some unmanned missions. The acceleration limit of 15g is more in keeping with solid rocket launch vehicles. Although high accelerations may require more structural mass, throttling the engines can significantly reduce specific impulse which will also cause vehicle growth. Optimization of acceleration limits requires more detailed knowledge about the mission than are currently available, and are

beyond the scope of this effort. An example POST input file is included in Appendix B.

POST will simulate multiple variations on the trajectory, perturbing the independent variables until the objective function is optimized. The vehicle point design was considered to be sized correctly if the trajectory simulation found a maximum payload of 10,000 lb<sub>m</sub>, with a tolerance of 50 lb<sub>m</sub>. Plots of example trajectory output data can be found in Appendix C.

## Chapter 5

### SSTO LAUNCH VEHICLE SIZE TRENDS

The gross mass and dry mass data obtained through the sizing process were used to produce graphs of the SSTO vehicle sizing trends, similar to the rough estimates in Figure 1 and the prior work by Stanley et al. presented in Figure 6.

#### Results for Near Term Structures with Minimal Insulation

Gross weight as a function of  $I_{sp}$  and bulk density for vehicles with near term structures and minimal insulation is presented in Figures 8, 9, and 10. Dry weight for these vehicles is presented in Figures 11, 12, and 13. A few additional data points not in the original design of experiments have been added to better illustrate trends.

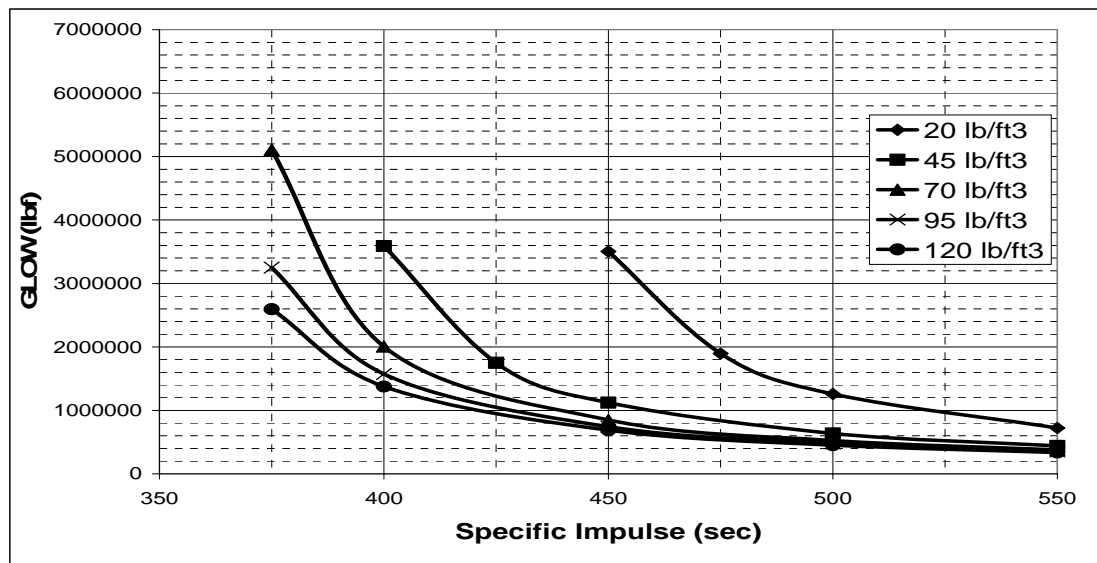


Figure 8. Gross Weight for SSTO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 150:1

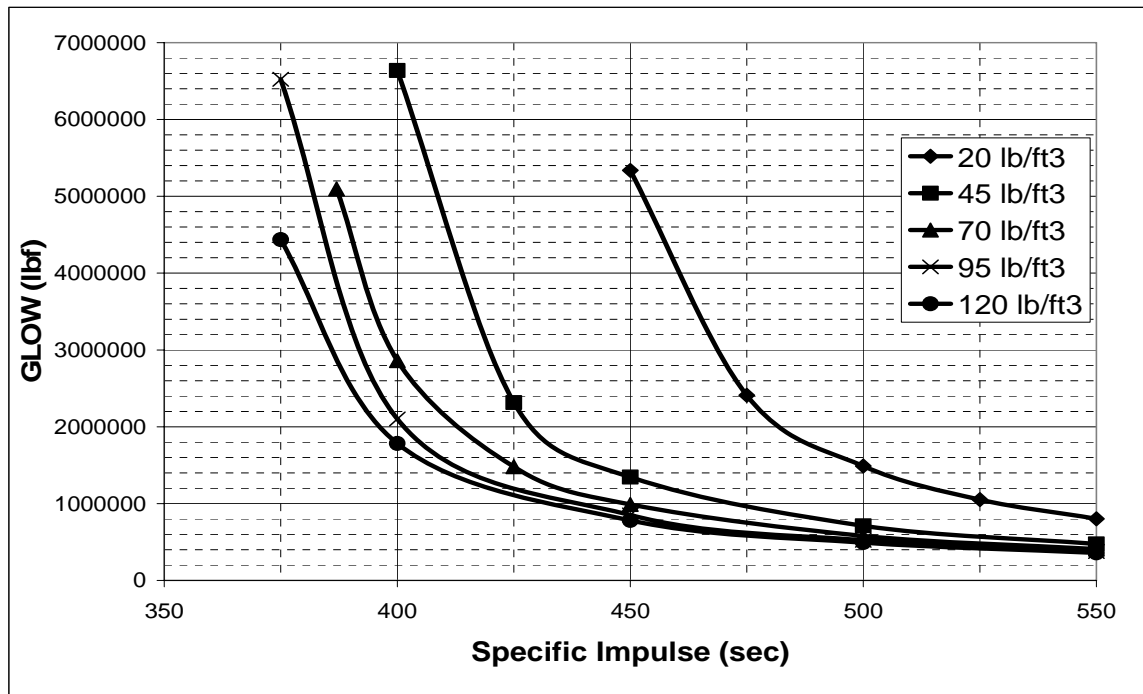


Figure 9. Gross Weight for SSTO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 100:1

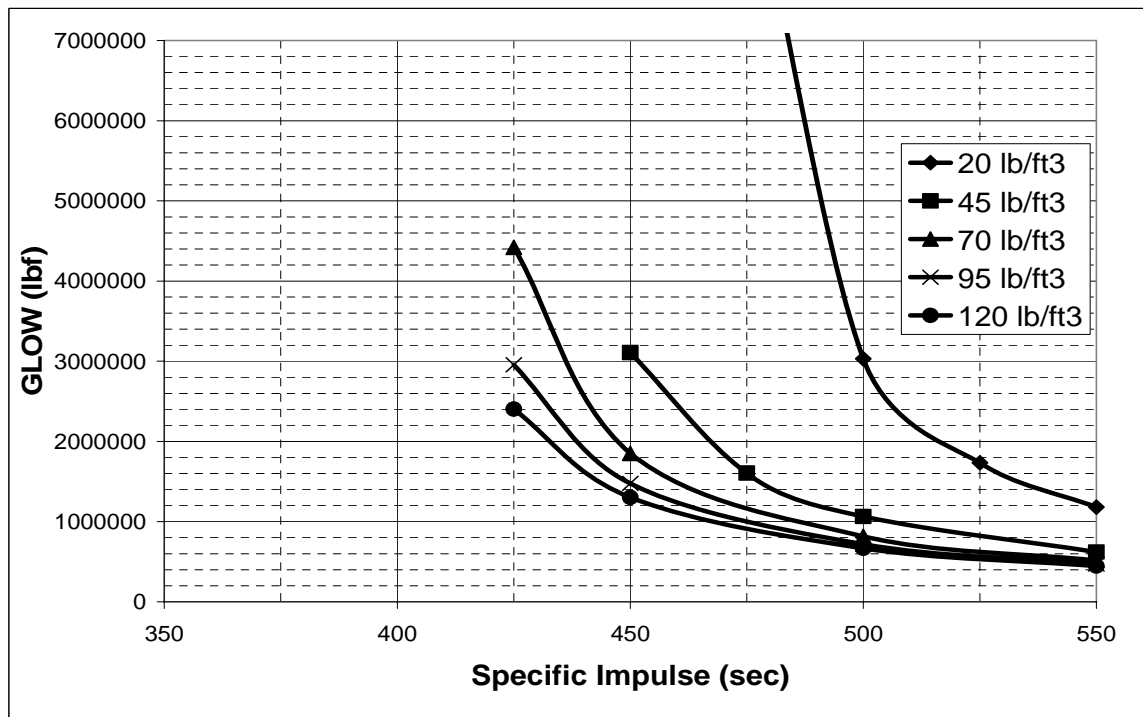


Figure 10. Gross Weight for SSTO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 50:1



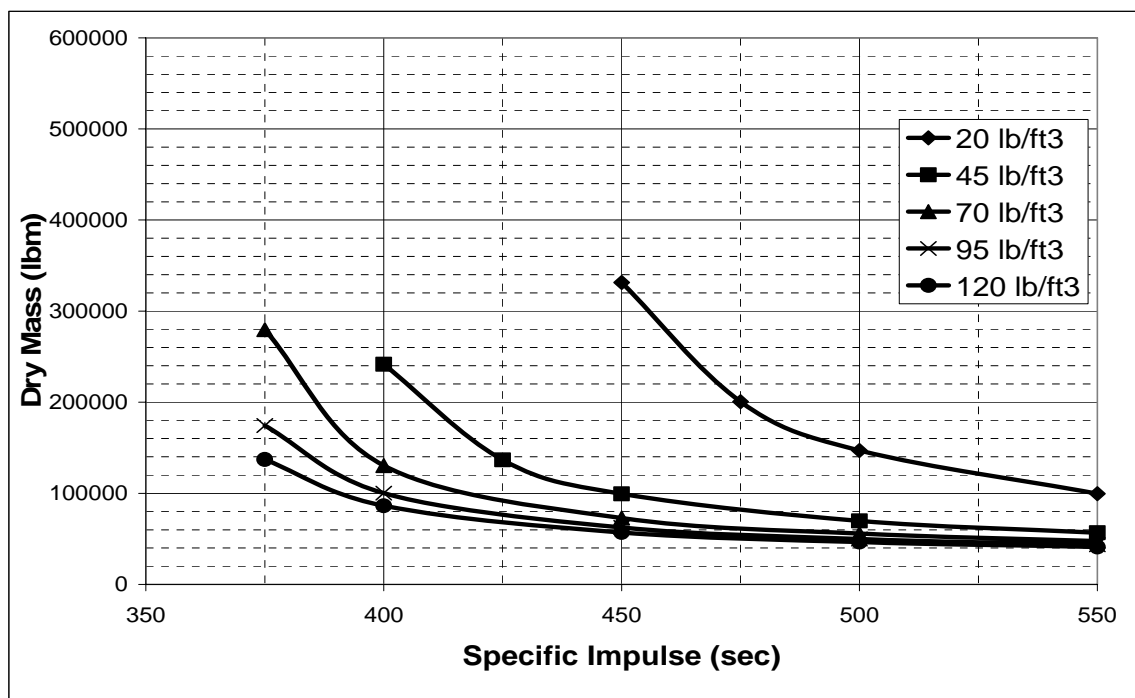


Figure 11. Dry Weight for SSTO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 150:1

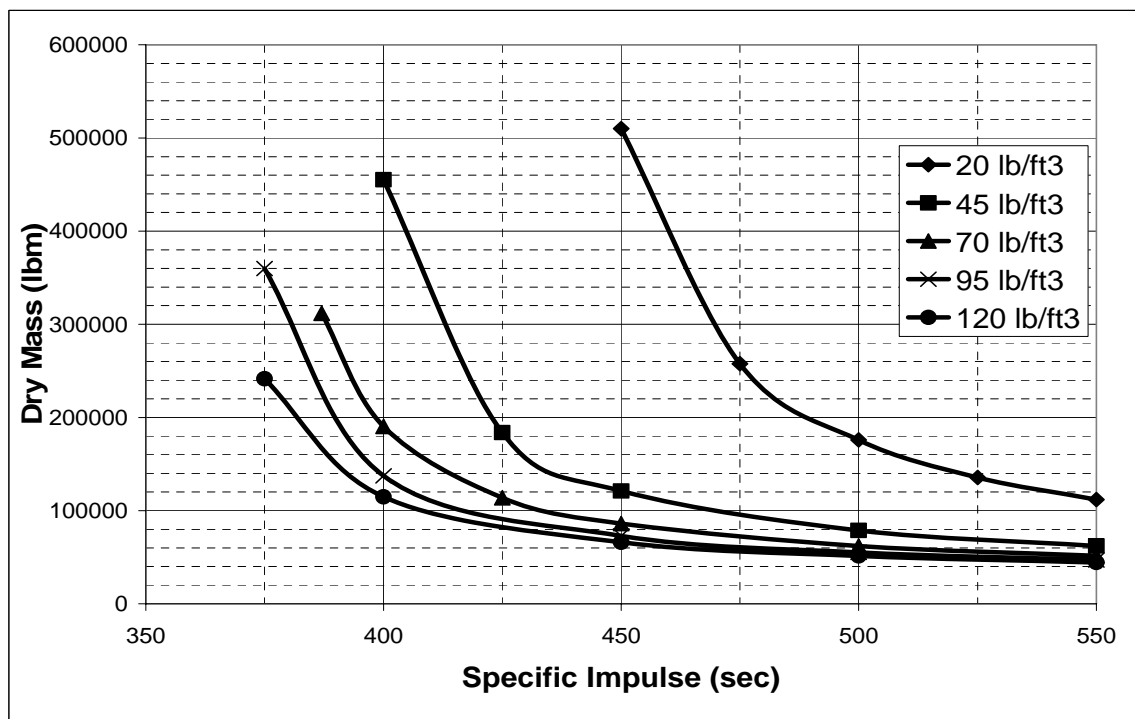


Figure 12. Dry Weight for SSTO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 100:1

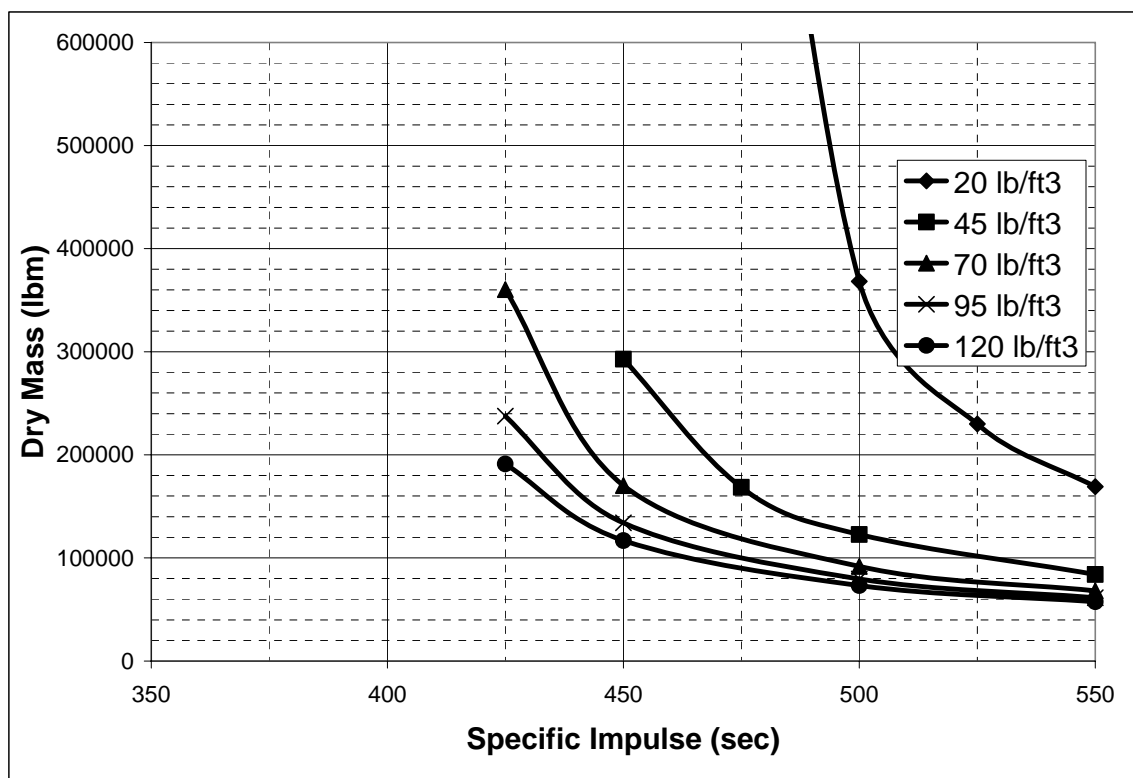


Figure 13. Dry Weight for SSTO Vehicles with Near Term Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 50:1

A few observations can be made from these figures. Overall, the trends follow the same shape as the curves in Figure 1 that used fixed mass fractions. Growth of an SSTO concept becomes extreme if the specific impulse falls below a certain value. SSTO size is also sensitive to bulk density, especially if the bulk density falls below 45 lb/ft<sup>3</sup>. Engine thrust to weight sensitivity is also evident, as designs that were feasible above a thrust to weight of 100:1 are unfeasible at 50:1.

#### Results for Near Term Structures with Heavy Insulation

Gross weight for vehicles with near term structures and heavy insulation are presented in Figures 14, 15, and 16. Dry weight for these vehicles is presented in Figures 17, 18, and 19.

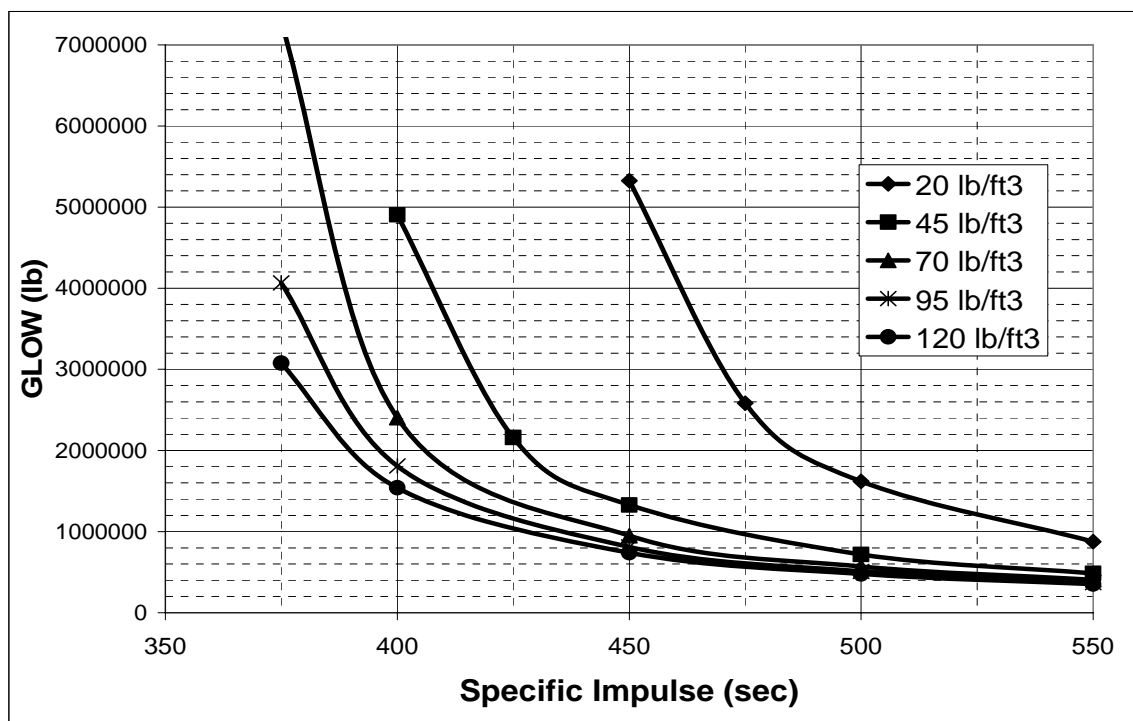


Figure 14. Gross Weight for SSTO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 150:1

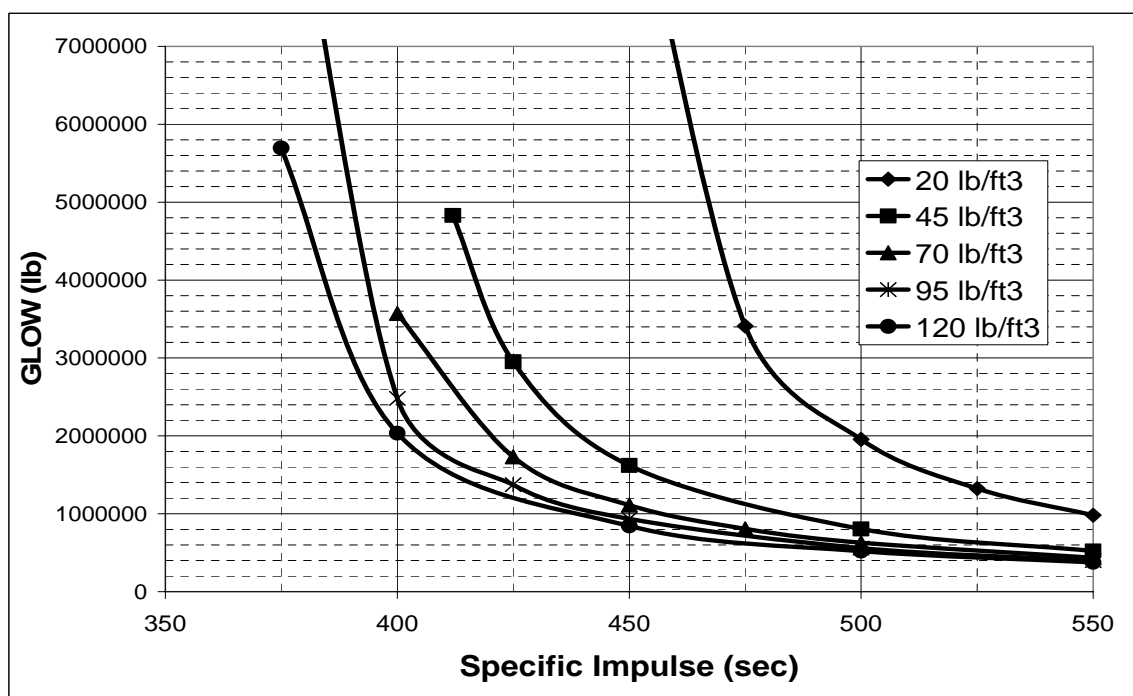


Figure 15. Gross Weight for SSTO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 100:1

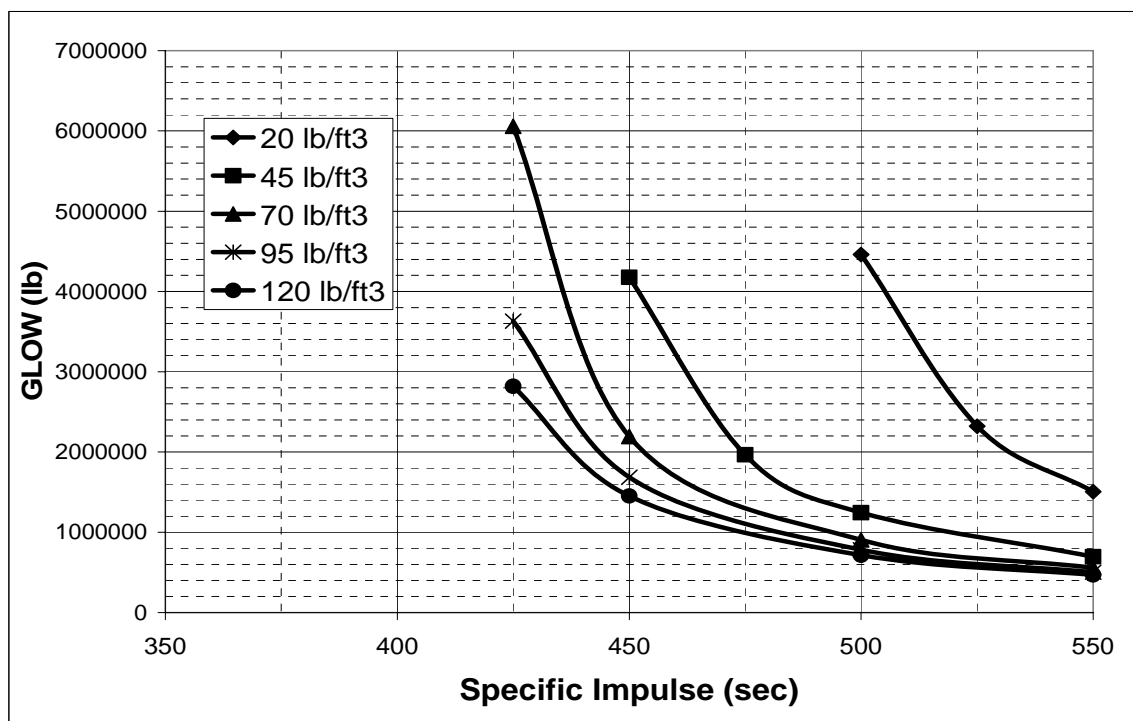


Figure 16. Gross Weight for SSTO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 50:1

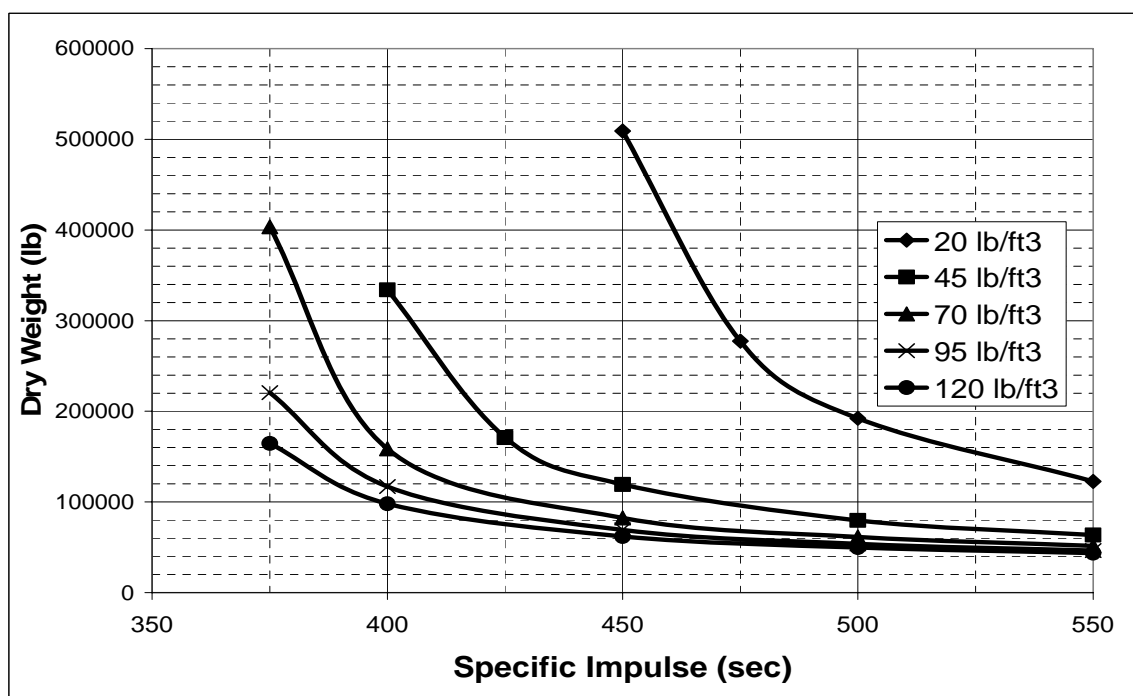


Figure 17. Dry Weight for SSTO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 150:1

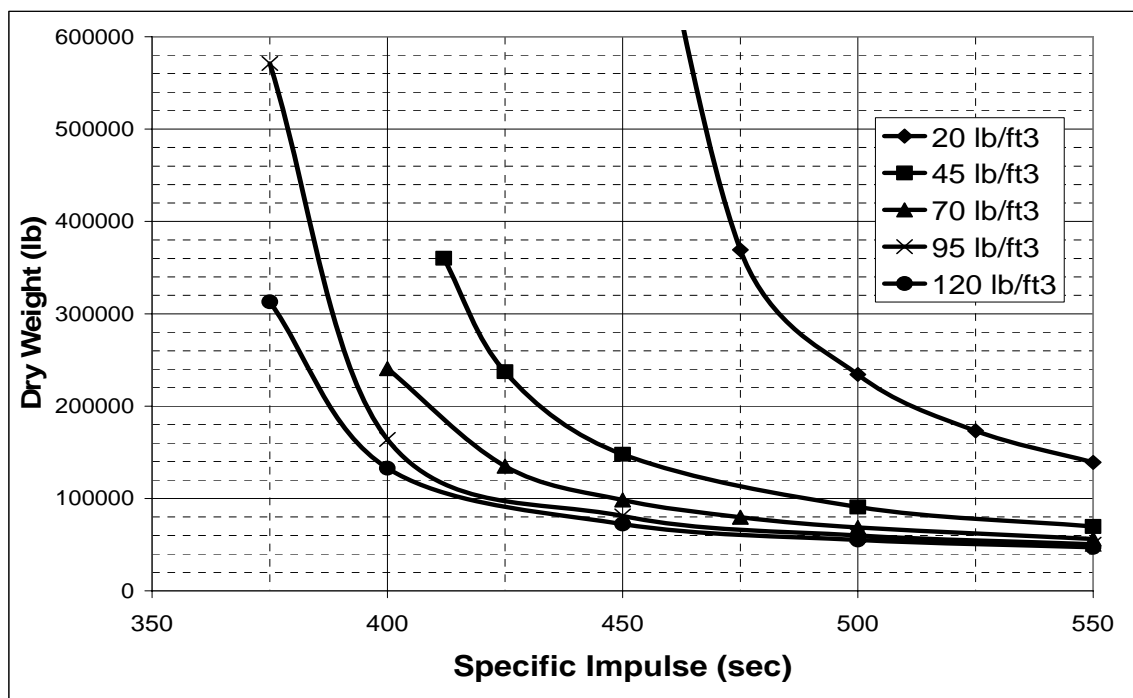


Figure 18. Dry Weight for SSTO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 100:1

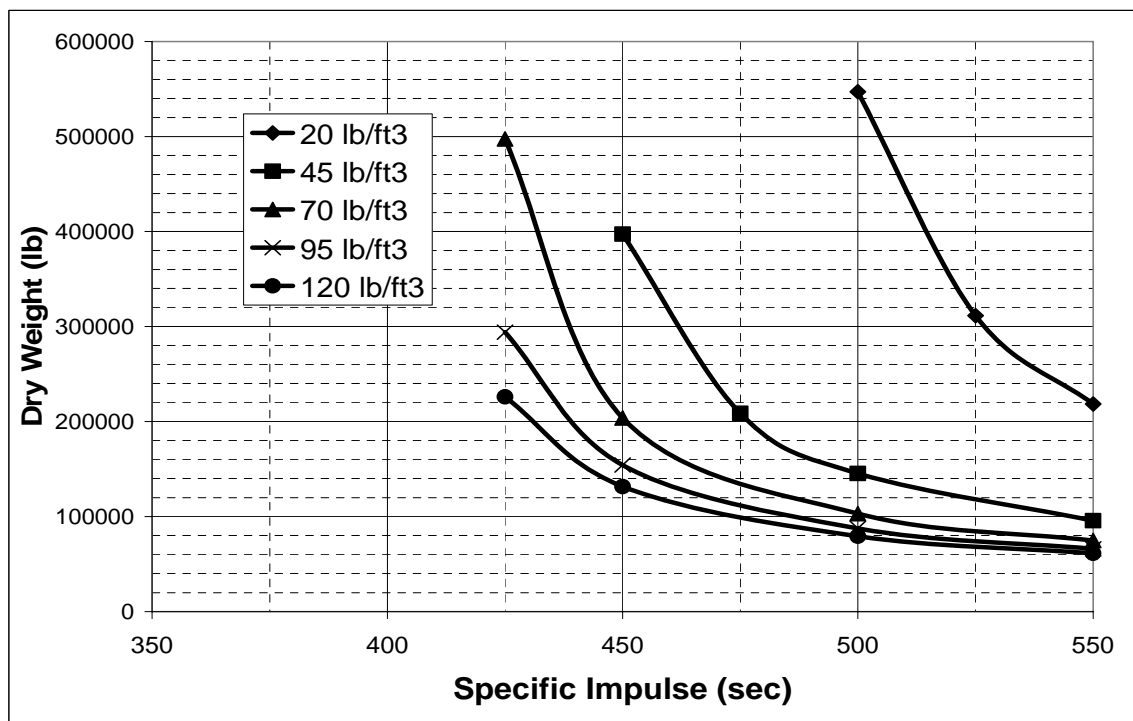


Figure 19. Dry Weight for SSTO Vehicles with Near Term Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 50:1

Observations are similar for the vehicles with heavy cryogenic insulation as for those with less insulation. The size of the vehicles is larger for the heavier insulation, as expected. The effect is most pronounced for vehicles that were already on the high slope / high growth factor part of the curves.

As LOX/LH<sub>2</sub> vehicles require heavy insulation over the majority of the tank surface, this set of curves may be the most instructive for evaluating LOX/LH<sub>2</sub> SSTO with metallic tanks. Trends on Figure 17 and 18 are in rough agreement with those of Stanley et al. on Figure 6. It is clear from these charts that a LOX/LH<sub>2</sub> SSTO vehicle would have a very high growth factor if near-term technologies were used, and that very high engine thrust-to-weight ratio would be required, as growth factor is high even when thrust-to-weight is 150:1.

#### Results for Advanced Structures with Minimal Insulation

Gross weight trends for vehicles with advanced structures and minimal insulation are presented in Figures 20, 21, and 22. Dry weight trends for these vehicles are presented in Figures 23, 24, and 25. The vehicle weights are significantly lower than for near term structures, illustrating the importance of advanced structures in enabling SSTO concepts. This set of charts represents the most optimistic assumptions used in the study.

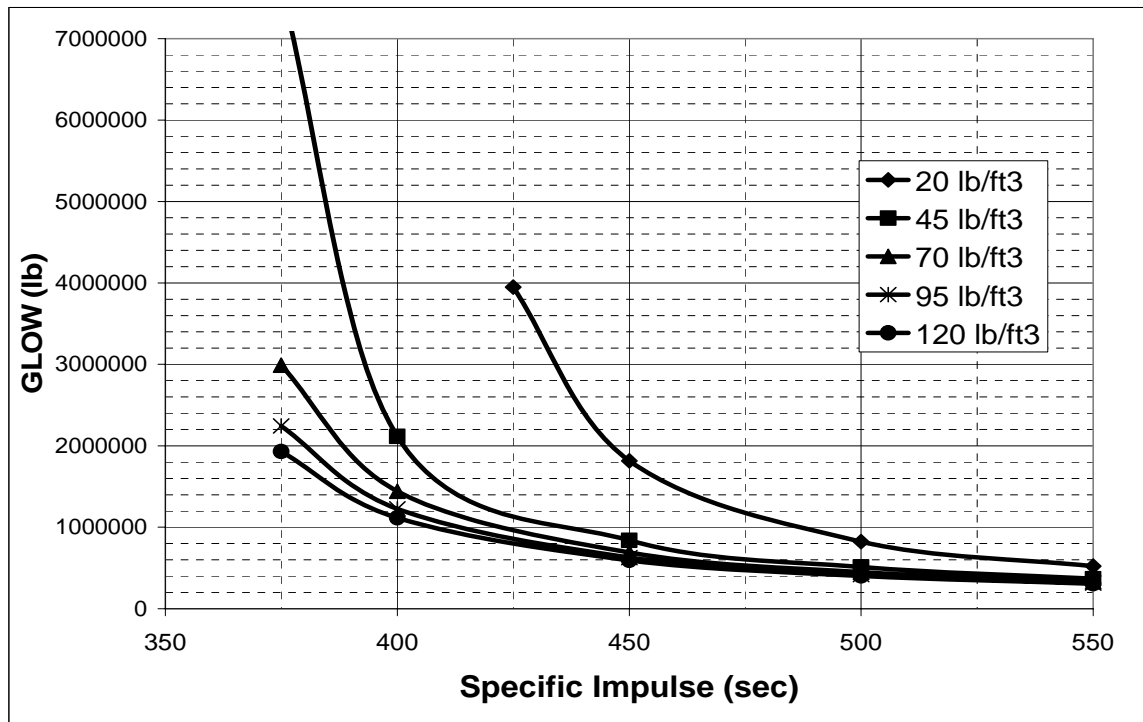


Figure 20. Gross Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 150:1

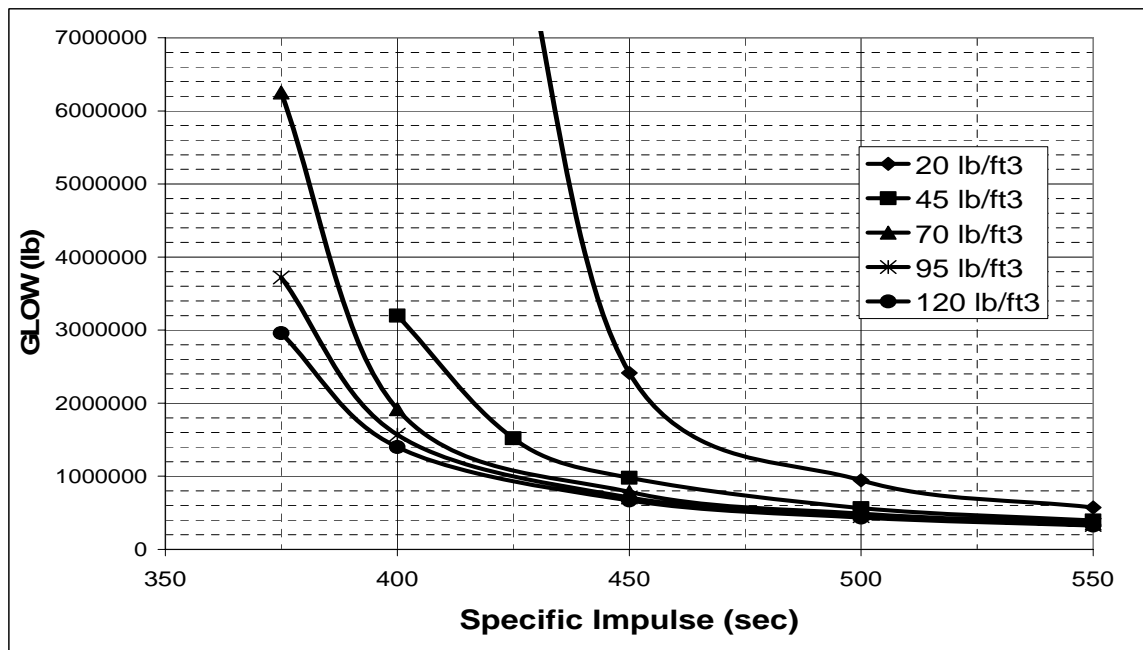


Figure 21. Gross Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 100:1

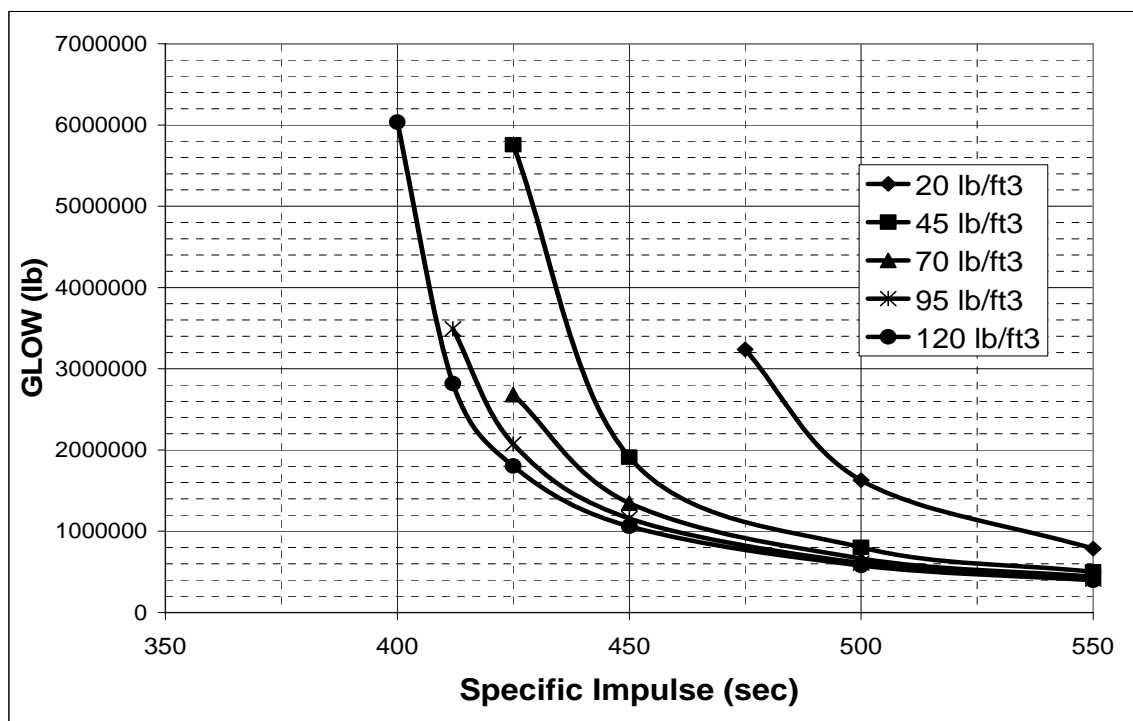


Figure 22. Gross Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 50:1

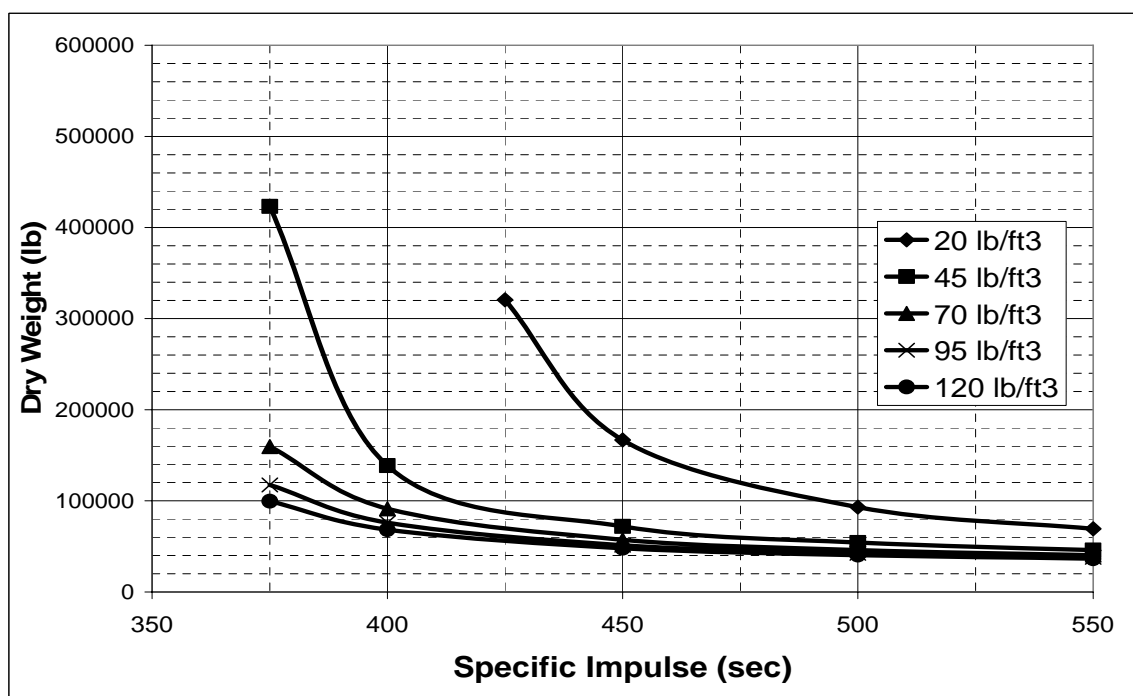


Figure 23. Dry Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 150:1



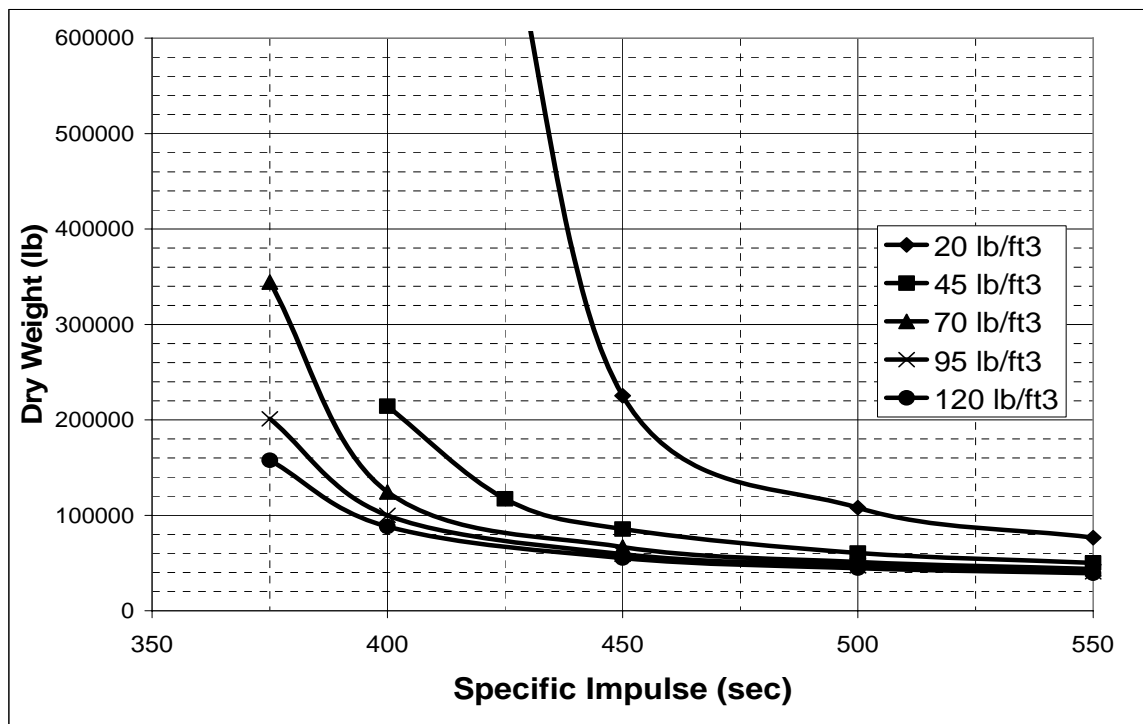


Figure 24. Dry Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 100:1

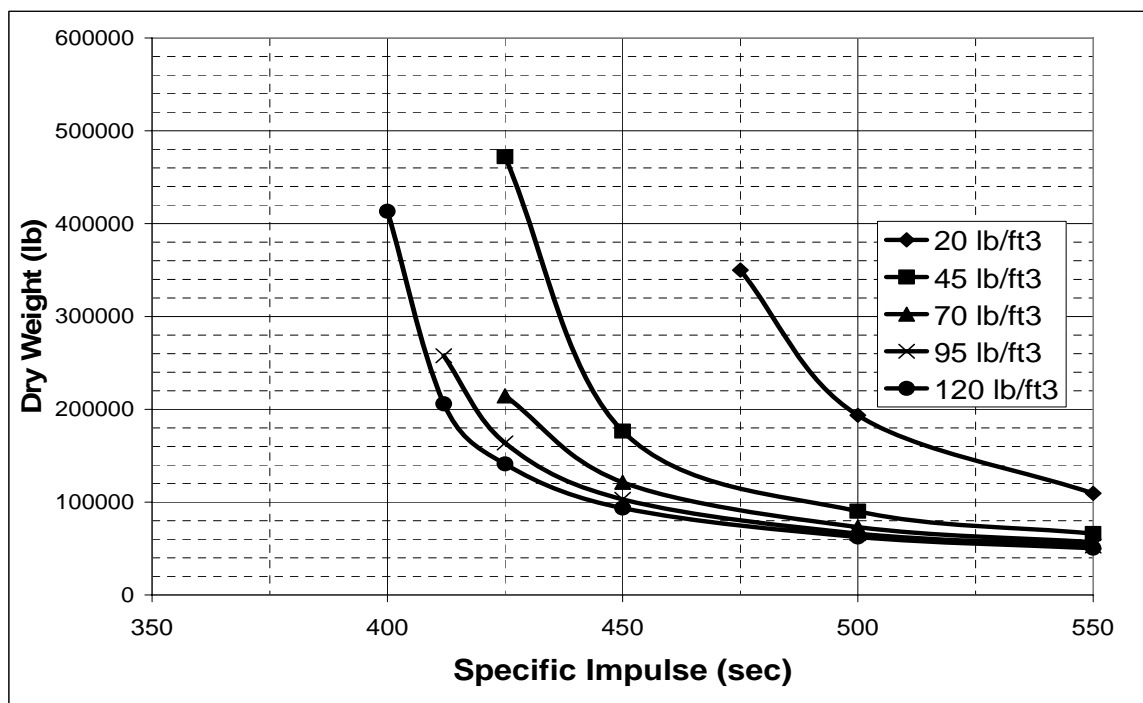


Figure 25. Dry Weight for SSTO Vehicles with Advanced Structures, Minimal Insulation, and Engine Thrust to Weight Ratio of 50:1

Results for Advanced Structures  
with Heavy Insulation

Gross weight trends for vehicles with advanced structures and minimal insulation are presented in Figures 26, 27, and 28. Dry weight trends for these vehicles are presented in Figures 29, 30, and 31. Vehicles with extremely low bulk density propellants are still significantly larger than vehicles with dense propellants. With advanced structures, a LOX/LH<sub>2</sub> SSTO would have fairly high growth factor, and would likely require engine thrust-to-weight greater than 100:1.

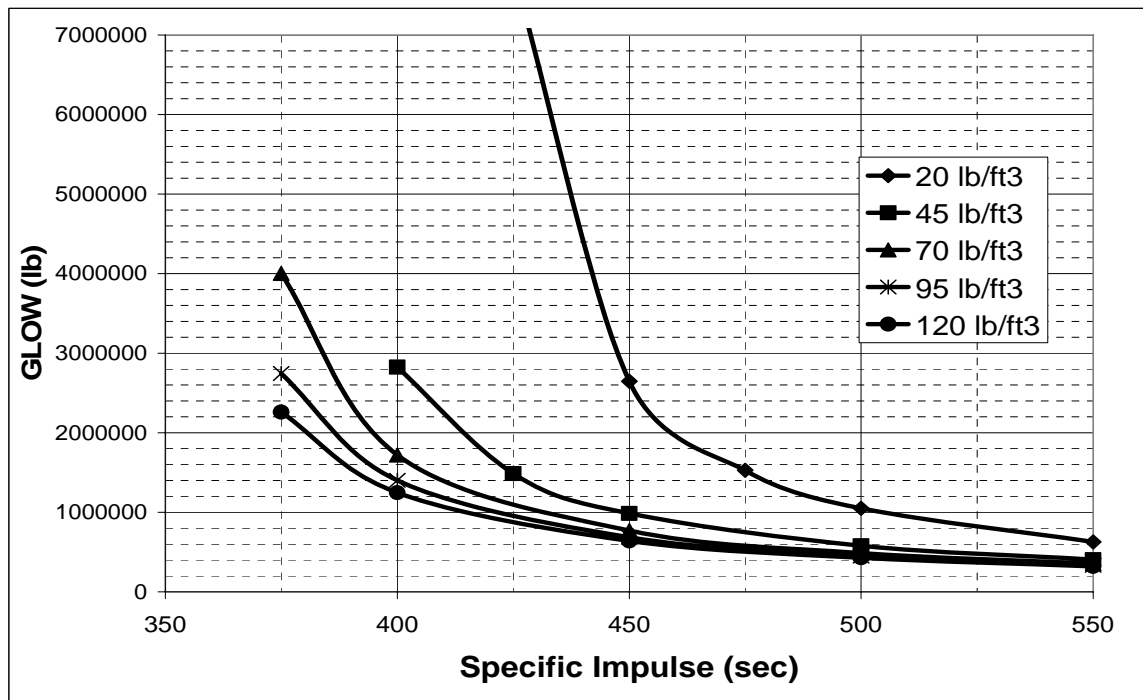


Figure 26. Gross Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 150:1

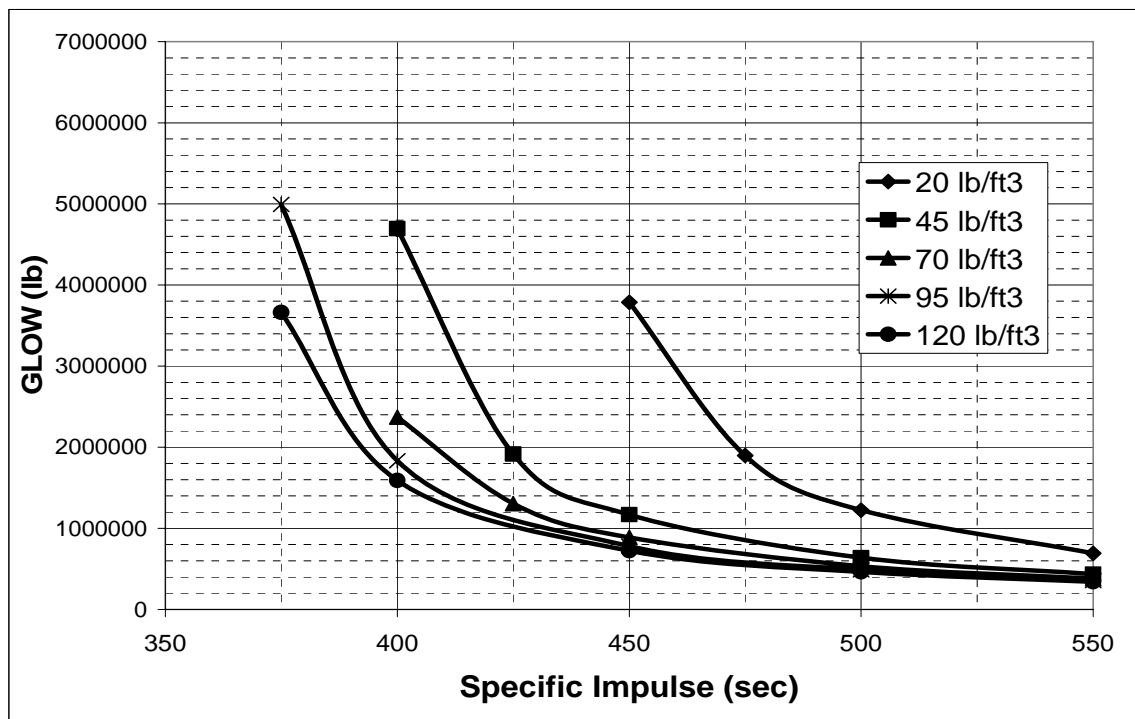


Figure 27. Gross Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 100:1

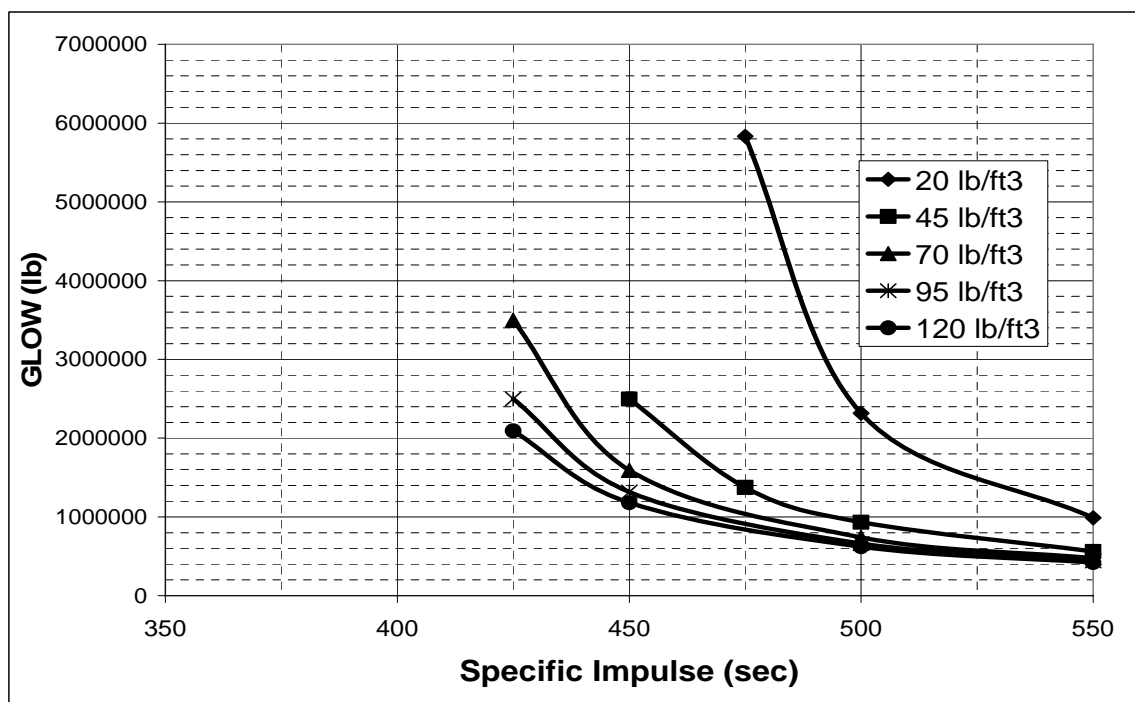


Figure 28. Gross Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 50:1

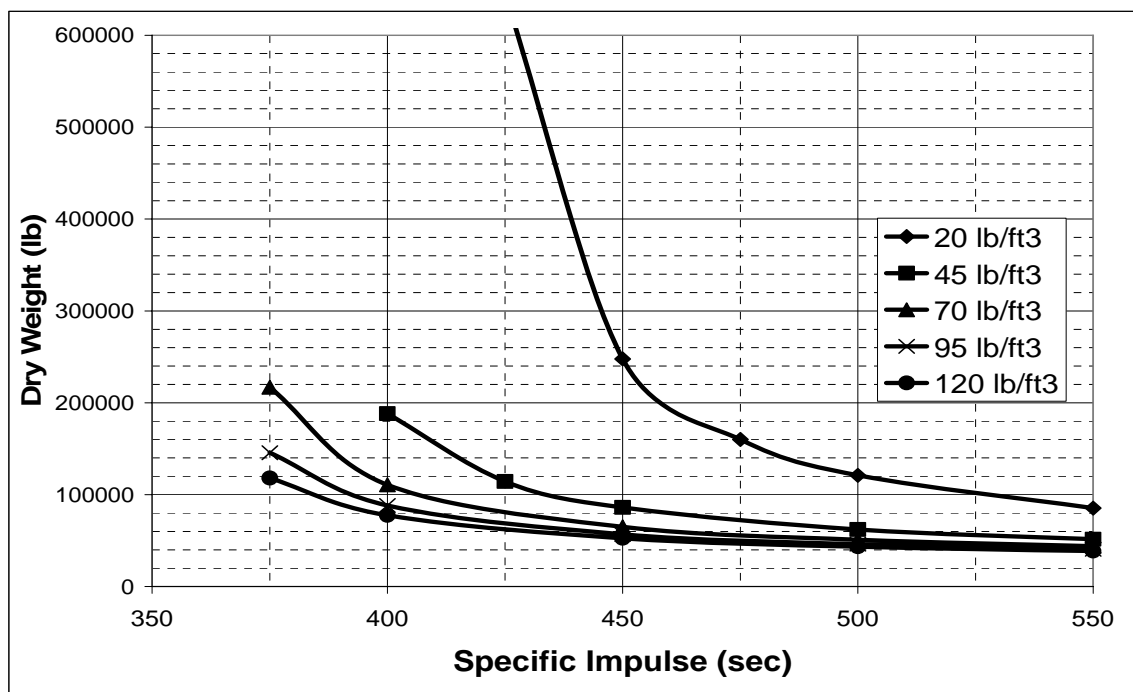


Figure 29. Dry Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 150:1

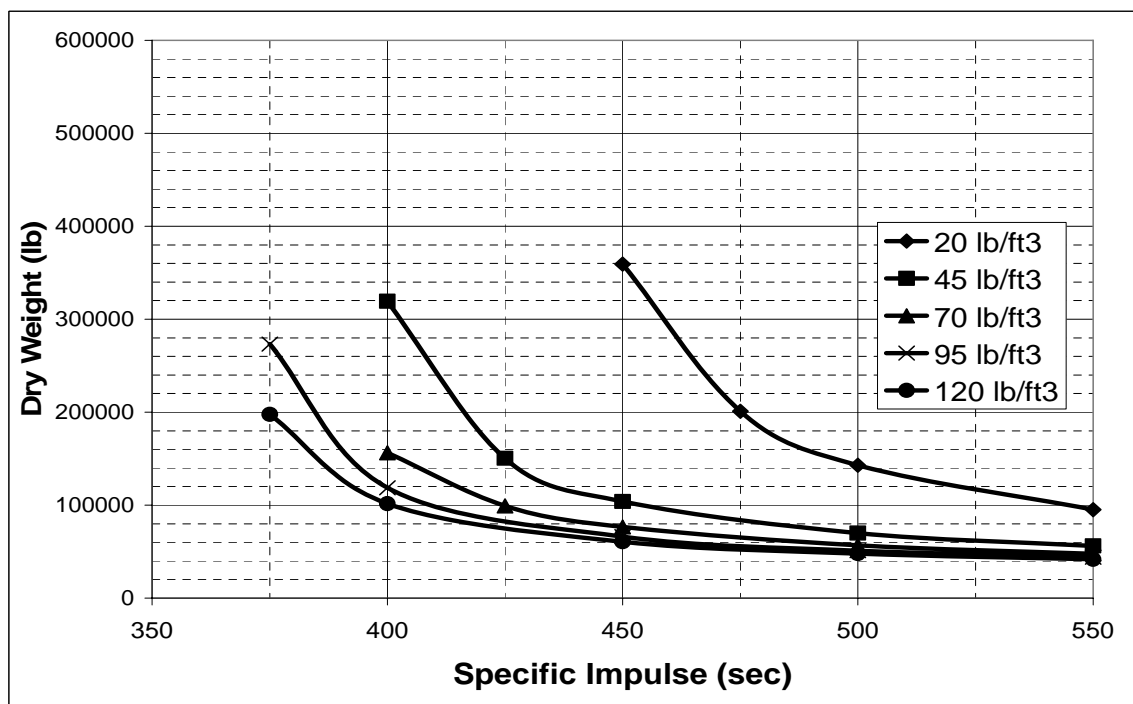


Figure 30. Dry Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 100:1

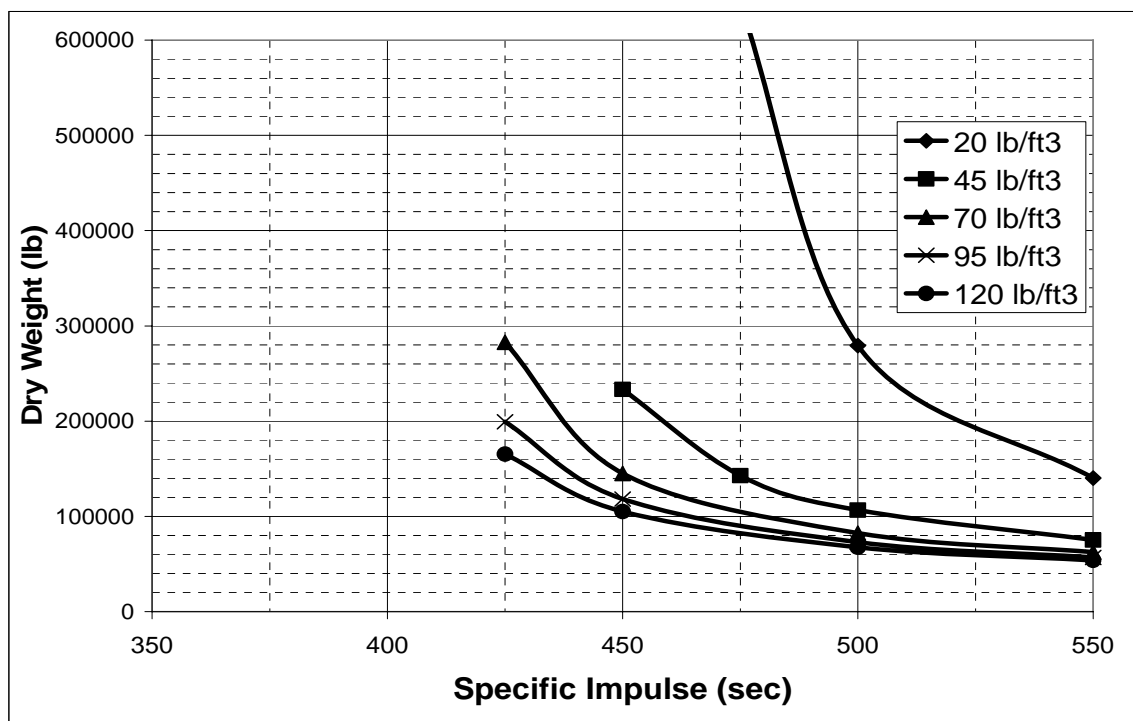


Figure 31. Dry Weight for SSTO Vehicles with Advanced Structures, Heavy Insulation, and Engine Thrust to Weight Ratio of 50:1

## Chapter 6

## CONCLUSIONS

A parametric study of 300 SSTO launch vehicle point designs was conducted to assess the impact of advanced propellant technologies on the feasibility of SSTO. Specific impulse, bulk density, and engine thrust to weight ratio were varied for vehicles with near term and far term structural technologies, both with and without insulation for deep cryogenic propellants. Vehicles were sized to deliver 10,000 lb<sub>m</sub> to a polar, once-around orbit, representative of potential future military reusable launch vehicle applications. A spreadsheet-based sizing tool was created based on mass estimating relationships in the Air Force Weights Analysis Tool. Vehicle sizes were confirmed through trajectory simulation with the Program to Optimize Simulated Trajectories.

Feasibility of SSTO with LOX/LH<sub>2</sub>

Results suggest that practical SSTO vehicles using LOX/LH<sub>2</sub> are not possible with current technologies. Any such vehicle would require at least a modest improvement in engine thrust to weight ratio, and would still possess extremely high growth factor.

Vehicle size and growth factor can be significantly reduced with the introduction of advanced structural technologies, such as composite propellant tanks. The vehicle will still be sensitive to engine thrust to weight ratio, and the growth factor will still be relatively high. Although vehicle designs are quite sensitive to improvements in bulk density, relatively small improvements that may be possible from hydrogen densification schemes<sup>71</sup> will not result in enough of an increase to enable SSTO vehicles. There are also impacts of reducing the temperature of liquid hydrogen to achieve the higher density, including increased

complexity of ground equipment and heavier tank insulation, which make this approach unattractive.

### Feasibility of SSTD with Advanced Propellants

Assuming that SSTD vehicles with gross mass below 1.5 Mlb<sub>m</sub> [680,000 kg] have growth factors low enough to be considered acceptable, Figure 32 shows the approximate region in the specific impulse-bulk density plane where such a vehicle can be enabled, assuming advanced structures, heavy cryogenic insulation, and a thrust to weight ratio of 150:1.

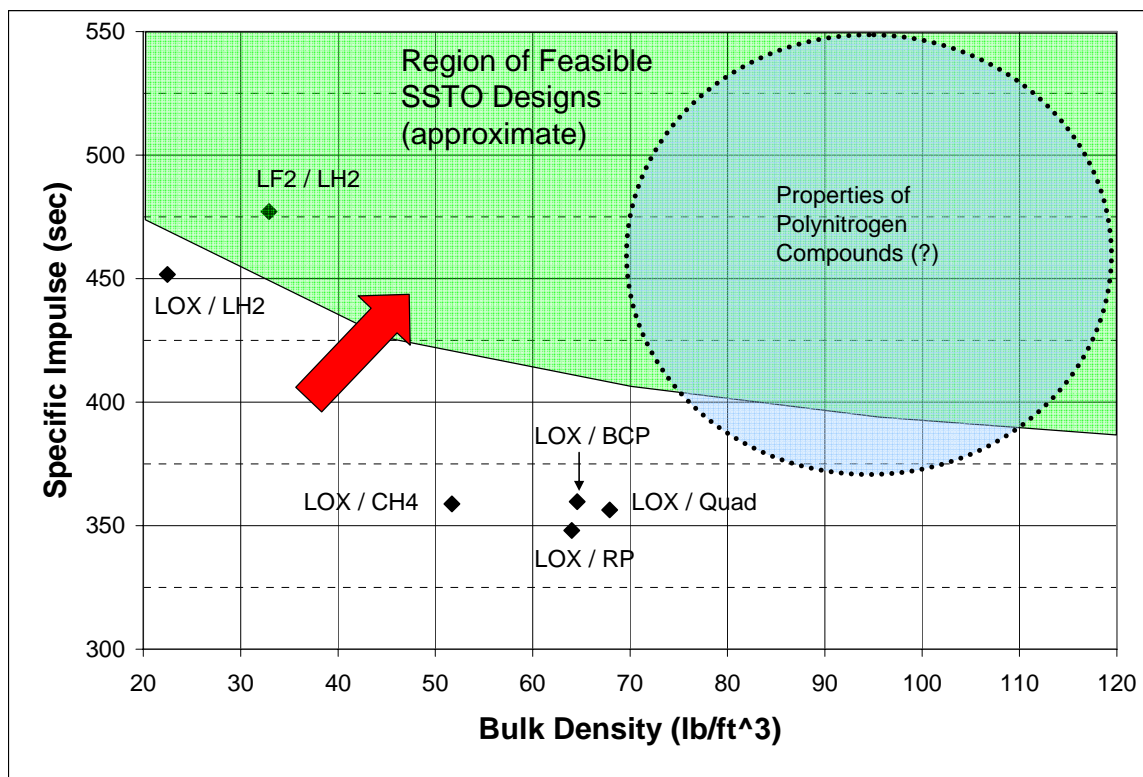


Figure 32. Region of SSTD Designs with GLOW less than 1.5 Mlb<sub>m</sub>

The arrow represents the general direction of technology thrust required to move towards SSTD vehicles. From the figure, it is clear to see why LOX/LH<sub>2</sub> is the propellant chosen for most SSTD studies. Of available propellants, it is the

closest to enabling SSTO, save  $\text{LF}_2/\text{LH}_2$ , which has unacceptable hazards. Hydrocarbon fuels, even the strained ring variety, do not provide enough specific impulse when burned with LOX to make a feasible SSTO. The fluorine-lithium-hydrogen tripropellant and other similar approaches will have even better performance to  $\text{LF}_2/\text{LH}_2$  but even greater obstacles to operability. No suitable polynitrogen propellants have yet been synthesized, but such a compound could offer the required performance to enable SSTO vehicles, provided it has the required characteristic for operability as well as performance.

#### SSTO versus TSTO for Economical RLVs

Even after technology has advanced to the required level to enable an SSTO vehicle, a TSTO vehicle may still be more cost effective. McClure and Andrews, as well as Eldred et al. reason that until technology has advanced to the point that TSTOs and SSTOs have similar growth factors (to the far right in Figure 1), SSTO vehicles will continue to emphasize low weight and maximum performance in subsystem design, as opposed to the TSTO emphasis on design for operability.<sup>72,73</sup> Since the original rationale for developing SSTO hardware was to reduce cost by simplifying operations, it makes little sense to build an SSTO using subsystems with poor operability as opposed to a TSTO using subsystems with improved operability. For that reason, any near term investment in RLVs would be best focused on operability improvements for TSTO vehicles, technologies that would still be applicable in the future when the difference in growth factor between SSTO and TSTO concepts is small enough to make SSTO vehicles an economical alternative.



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## APPENDICES

APPENDIX A  
EXAMPLE SSTD VEHICLE MASS ESTIMATE



**SSTO Sample Point Design Weight Estimate**  
**Advanced Vehicle Structures**  
**Heavy Cryogenic Insulation**  
**Specific Impulse = 400 sec**  
**Bulk Density = 120 lb/ft<sup>3</sup>**  
**Engine Vacuum T/W = 100**

		<b>Mass (lb<sub>m</sub>)</b>
<b>1.0.0</b>	<b>Aero Surfaces</b>	<b>9539</b>
1.1.0	Exposed Wing	7469
1.2.0	Vertical Tail	1198
1.3.0	Elevons	536
1.4.0	Body Flap	336
<b>2.0.0</b>	<b>Body Structure</b>	<b>5886</b>
2.1.0	Nose Structure	384
2.2.0	Intertank Structure	807
2.3.0	Thrust Structure Cone	4592
2.4.0	Aft Body/Engine Fairings/Wing Fairings	103
<b>3.0.0</b>	<b>Thermal Protection System</b>	<b>5473</b>
3.1.0	Internal Fuselage Insulation	210
3.2.0	External TPS	5263
3.2.1	Nose & Wing/Cntrl Surface Leading Edge	774
3.2.2	Underbody and Underwing	2752
3.2.3	Top Fuselage and Wing	1193
3.2.4	Tail	442
3.2.5	Body Flap	87
3.2.6	Landing Gear Bay	15
<b>4.0.0</b>	<b>Takeoff/Landing Gear</b>	<b>3642</b>
4.1.0	Running Gear	2049
4.2.0	Gear Structure	1593
4.2.1	Control/Mechanism and Actuator	1593
<b>5.0.0</b>	<b>Rocket Engine</b>	<b>22321</b>
<b>6.0.0</b>	<b>Orientation Control System</b>	<b>5382</b>
6.1.0	Aerodynamic Controls	1744
6.2.0	RCS (Reaction Control System)	3638
6.2.1	Thrusters and Supports (Fwd)	715
6.2.2	Tanks	74
6.2.3	Lines, Manifolds, Regulators	633
6.2.4	Electric Pumps & Pressurant	78
6.2.5	RCS Fuel	822

6.2.6	RCS Oxidizer	1316
<b>7.0.0</b>	<b>Power Supplies</b>	<b>4890</b>
7.1.0	APU (Auxiliary Power Unit)	440
7.1.1	APU Fuel	91
7.1.2	APU Oxidizer	164
7.1.3	Plumbing, Valves, Lines	185
7.2.0	Other Power Supplies	1036
7.2.1	Fuel Cells	1036
7.3.0	Power Conversion and Distribution	3404
7.4.0	Fuel Control	10
<b>8.0.0</b>	<b>Propellant Tanks</b>	<b>20926</b>
8.1.0	Oxidizer, Main Tanks	7746
8.2.0	Fuel, Main Tanks	2105
8.3.0	Tank Structural Provisions	1839
8.4.0	Boost and Transfer Pumps	4480
8.5.0	Dump and Drain System	345
8.6.0	Main Tanks Pressurization System	3530
8.6.1	Cryogenic Propellant Pressurization Sys	3530
8.7.0	Purge Gas & System	881
<b>9.0.0</b>	<b>Propellant Stage Interconnects</b>	<b>0</b>
9.1.0	Propellant Stage Interconnects	0
<b>10.0.0</b>	<b>Avionics</b>	<b>6360</b>
10.1.0	Actuator Control	1137
10.2.0	Range Safety (no crew)	83
10.3.0	Audio/Video (crew only)	0
10.4.0	Guidance, Navigation & Control	936
10.5.0	Communications and Tracking	837
10.6.0	Displays and Control (crew only)	0
10.7.0	Instrumentation System	664
10.8.0	Equipment Cooling	174
10.9.0	Heat Transport System	223
10.10.0	Heat Rejection System	345
10.11.0	Flash Evaporator System	48
10.12.0	Data Processing	1317
10.13.0	Cabling	365
10.14.0	Structural Support/ Shelf	231
<b>11.0.0</b>	<b>Electronic Health Management</b>	<b>620</b>
11.1.0	Vehicle	620
<b>12.0.0</b>	<b>Hydraulic Power</b>	<b>395</b>
12.1.0	Hydraulic/Pneumatic System	19

12.2.0	Conversion and Distribution	376
<b>13.0.0</b>	<b>Growth Margin</b>	<b>13249</b>
13.1.0	Design Reserve/ Dry Weight Margins	13249
<b>14.0.0</b>	<b>Crew Provisions</b>	<b>0</b>
14.1.0	Crew Provisions/Waste Management	0
14.2.0	Crew Cabin	0
14.3.0	Environmental Control	0
14.3.1	ECLSS Fuel (Life Support System)	0
14.3.2	ECLSS Oxidizer (Life Support System)	0
14.4.0	Crew Canopy/Fairing	0
14.5.0	Crew	0
<b>15.0.0</b>	<b>Drag Chute</b>	<b>245</b>
15.1.0	Drag Chute	245
<b>16.0.0</b>	<b>Payload Bay Structure</b>	<b>2648</b>
16.1.0	Container	1208
16.2.0	Provisions	1440
	<b>Dry Weight Total</b>	<b>101575</b>
<b>17.0.0</b>	<b>Residual Propellant</b>	<b>29592</b>
17.1.0	Trapped fuel, Main Tanks	4227
17.2.0	Trapped oxidizer, Main Tanks	25364
<b>18.0.0</b>	<b>Payload</b>	<b>10000</b>
18.1.0	Payload Delivered	10000
	<b>Landing Weight Total</b>	<b>141167</b>
<b>19.0.0</b>	<b>Usable Main Propellant</b>	<b>1450000</b>
19.1.0	Fuel	207143
19.1.1	Oxidizer	1242857
	<b>Gross Lift Off Weight</b>	<b>1591167</b>
	<b>Stage Usable Propellant Mass Fraction</b>	<b>0.917</b>

APPENDIX B  
EXAMPLE POST INPUT FILE

```

P$SEARCH
C*****
C Parametric SSTO Study
C Summer 2006
C 400 sec Isp
C 120 lb/ft3 propellant bulk density
C 100:1 engine T/W
C Heavy cryogenic insulation
C
C LIGHTWEIGHT COMPOSITE TANKS
C
C Optimize for maximum Payload
C
C
C      15g acceleration limit
C
C
C Jason Mossman
C California State University, Fresno
C*****
C
C      SRCHM      = 4,          / unaccelerated projected gradient
C*** optimization variable ***
C      OPTVAR     = 'WEIGHT',   / optimization variable is dot product downrange
C      OPT        = 1,          / maximize optimization variable
C      OPTPH      = 100,        / maximize dwnrng at event 100
C      WOPT       = 1.00E-006,   / optimization weighting factor = 1/optvar
value
C      PCTCC      = 1.0,        / max percent chg in mag of control vector, u
C      MAXITR     = 30,        / maximum number of iterations
C      IPRO       = -1,        / print final trajectory
C      IOFLAG     = 0,        / English units for inputs/outputs
C*** control variables - 13 pitch rates, and payload weight***
C      NINDV      = 14,
C      INDVR      = 13*'PITPC2', 'WPLD',
C      INDPH(1)   = 100, 120, 125, 130, 135,
C                  136, 137, 138, 138, 140,
C                  145, 150, 151,
C                  1,
C
C      u = 6.62201865894866630e-001, -8.77013574867444470e-001,
C      -5.75943383647670230e-006, -1.67393180811082050e+000,
C      -3.39508551696323210e-001, -2.38675579107067330e-001,
C      -3.21588498118405510e-001, -4.41167076825992810e-002,
C      -4.71397243816572440e-001, -6.41723865244188120e-002,
C      -1.01768652399518020e+000, -3.95715807573182310e-001,
C      -4.52244443873128010e-001, 9.99177073368263520e+003,
C
C*** constraint variables - max q, max/min alpha, flight path angle,
velocity & altitude at burnout***
C      NDEPV      = 6,
C      DEPVR      = 'XMAX1', 'XMIN2', 'XMAX2', 'GAMMAI', 'VELI', 'ALTITO',
C      DEPVAL     = 700., -5.0, 5.0, 0., 25841., 340495.,
C      DEPPH      = 190, 130, 130, 190, 190, 190,
C      DEPTL      = 1.0, 0.1, 0.1, 0.005, 10., 100.,

```

```

      IDEPVR = 1,      -1,      1,      0,      0,      0,
$
P$GENDAT
  TITLE = 'Parametric SSTO Vehicle',
  EVENT = 1,
  FESN = 190,        / final event number
  NPC(2) = 1,        / 4th order runge-kutta integration
    DT = 0.5,        / integration step size, sec
    PINC = 10,       / print interval (no. of integration steps)
    PRNC = 5,        / print interval to PROFIL (no. of integration
steps)
C*** initial conditions ***
  NPC(3) = 4,        / initial velocity vector in earth local
horizontal frame
  VELR = 0,          / zero initial velocity
  GAMMAR = 89.99,    / vertical launch (90 deg flight path angle)
  AZVELR = 0.0,      / initial azimuth of relative velocity vector
  NPC(4) = 2,        / initial position in spherical (lat, long,
alt) coordinates
  altito = 0.,       / geodetic altitude for VAFB, ft
  GDLAT = 34.7,      / geodetic latitude for VAFB, deg
  AZL = 180.0,       / 0 deg (due North) azimuth for launch
coordinate system
  LONG = 239.0,      / relative longitude for VAFB, deg E
  NPC(5) = 5,        / 1976 standard atmosphere
  NPC(12) = 2,       / compute downrange/crossrange relative to
inertial great circle
  AZREF = 180.0,     / 0 deg reference azimuth for
downrange/crossrange calculations
  LONREF = 239.0,    / reference longitude for range calculations
  NPC(16) = 0,       / oblate earth gravity model (1 for spherical
gravity model)
  NPC(19) = 1,       / print input conditions for each phase
  NPC(25) = 3,       / compute velocity losses and output at each
print interval
  NPC(28) = 0,       / no tracking station calculations
  NPC(7) = 1,        / throttle to limit acceleration
  ASMAX = 15.0,      / acceleration limit
  NPC(8) = 2,        / axial/normal aerodynamic coefficients
  SREF = 2763.,      / aerodynamic reference area for Stage 1, ft2
  NPC(9) = 1,        / rocket engine
  NENG = 1,          / total number of engines on vehicle
  IENGMF = 1,        / Stage 1 Engine is on and is a Rocket
  IWDF(1) = 2,       / Calc Flowrate as function of Thrust and Isp
  NPC(30) = 0,       / N-stage composite weight calculation model
  MONX(1) = 'dynp','alpha','asmg', / monitor variables
  ISPV(1) = 400.,    / Stage 1 Isp
  WGTSG = 1581167., / Booster weight
  WPROPI = 1450000.0, / Stage 1 Expelled Propellant Wt
  WPLD = 10000.,    / payload;
C *** Guidance Inputs ***
  IGUID(1) = 1,      / inertial euler angles
  IGUID(4) = 1,      / cubic polynomial, input pitpc, rolpc, yawpc
    PITPC = 0.0,
    ROLPC = 0.0,

```

```

    YAWPC = 0.0,
    ALTMIN  = -3.E4,      / minimum altitude
    ALTMAX  = 1E8,        / maximum altitude
    MAXTIM  = 6.0E3,      / maximum time
    PRNT(91) = 'TVAC','XMAX1','XMAX2','XMIN2','XMAX3','PSTOP', /
additional print variables
$
P$TBLMLT
    TVC1M  = 1.0,
    TVC2M  = 1.0,
    TVC3M  = 1.0,
    CDM     = 1.0,      / table multipliers
$
P$TAB
C  Vac Delivered Thrust for Stage 1 Motor:
    TABLE = 'TVC1T', 0, 2232138.,/

C  Total Exit Area for Stage 1 Motor:
    TABLE = 'AE1T', 0, 152.29, /
$
C *** STS Orbiter aerodynamics
*include 'C:/Documents and Settings/mossman jason/My Documents/Temp
Post/STS_Aero.dat'
$
P$GENDAT
    EVENT=100, CRITR= 'TIME', VALUE= 10.0, / initial vehicle pitchover,
see u-vector for commanded pitch polynomial coeff
    IGUID(4) = 0., / cubic polynomial, present program value
    ROLBD=180.0,
    ENDPHS=1,
$
P$GENDAT
    EVENT=120, CRITR = 'TIME', VALUE=15.0, /
    ENDPHS=1,
$
P$GENDAT
    EVENT=125, CRITR = 'TIME', VALUE=30.0, /
    ENDPHS=1,
$
P$GENDAT
    EVENT=130, CRITR = 'TIME', VALUE=45.0, /
    ENDPHS=1,
$
P$GENDAT
    EVENT=135, CRITR = 'TIME', VALUE=60.0, /
    ENDPHS  = 1,
P$GENDAT
    EVENT = 136, CRITR = 'TIME', VALUE = 90.0,
    ENDPHS = 1,
$
P$GENDAT
    EVENT = 137, CRITR = 'TIME', VALUE = 100.0,
    ENDPHS  = 1,
$

```

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P$GENDAT
  EVENT = 138, CRITR = 'TIME', VALUE = 130.0,
  ENDPHS = 1,
  $
P$GENDAT
  EVENT = 139, CRITR = 'TIME', VALUE = 150.0,
  ENDPHS=1,
  $
P$GENDAT
  EVENT=140, CRITR = 'TIME', VALUE=170.0, /
  ENDPHS=1,
  $
P$GENDAT
  EVENT=145, CRITR = 'TIME', VALUE=190.0,
  ENDPHS=1,
  $
P$GENDAT
  EVENT=150, CRITR = 'TIME', VALUE=200.0, /
  ENDPHS=1,
  $
P$GENDAT
  EVENT=151, CRITR = 'TIME', VALUE=210.0,
  NPC(1) = 2, / orbital parameters output print block ON
  ENDPHS=1,
  $
P$GENDAT
  EVENT=190, CRITR = 'WPROP', VALUE = 0.0, /
  TITLE   = '* -- Burnout -- *',
  IGUID(1) = 3,0,1, / inertial aerodynamic angles
C          use the same steering options for alpha, beta,
bank
C          cubic polynomial, input value
  ALPPC(1) = 0., /
  DT       = 1., / integration step size
  NPC(9)   = 1, / rocket engine
  IENGMF   = 0, / all engines are off
  ENDJOB   = 1,
  ENDPRB   = 1,
  $

```



APPENDIX C  
EXAMPLE TRAJECTORY OUTPUT DATA

Trajectory output data is presented for the vehicle with 400 sec.  $I_{sp}$ , 120 lb<sub>m</sub>/ft<sup>3</sup> propellant bulk density, 100:1 engine T/W, advanced structures, and heavy cryogenic insulation.

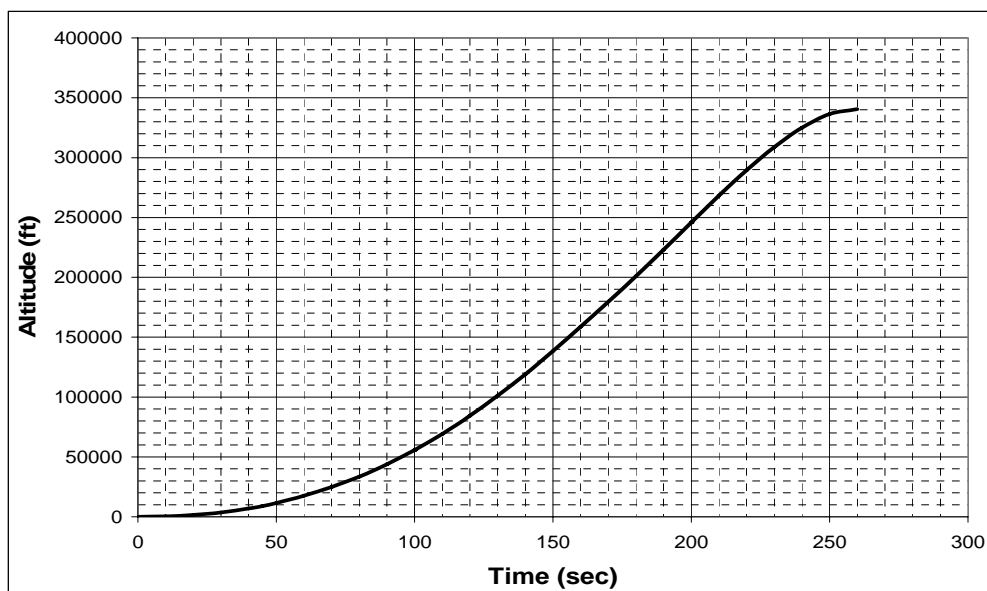


Figure C1. Altitude vs. Time for SSTO Vehicle

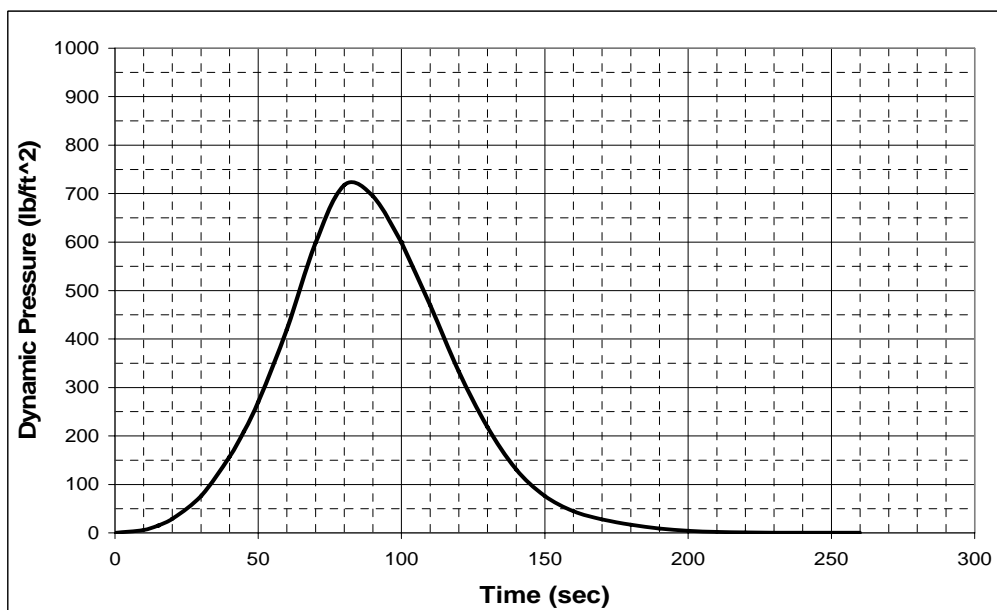


Figure C2. Dynamic Pressure vs. Time for SSTO Vehicle

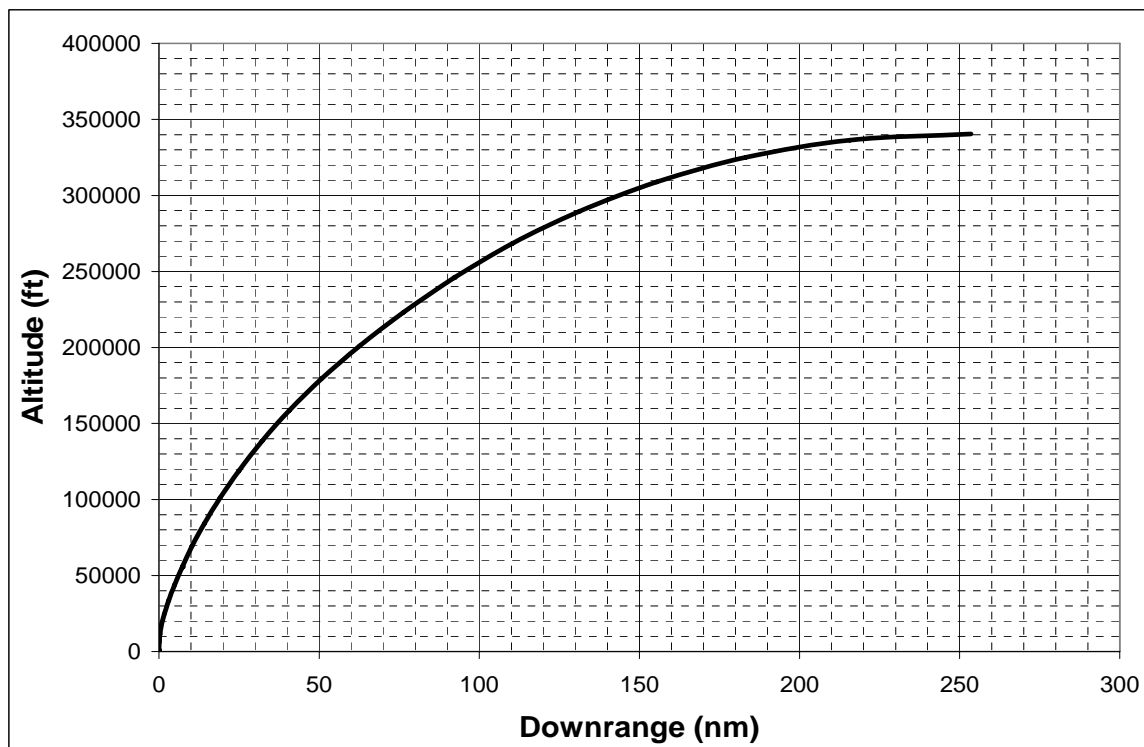


Figure C3. Altitude vs. Downrange for SSTO Vehicle

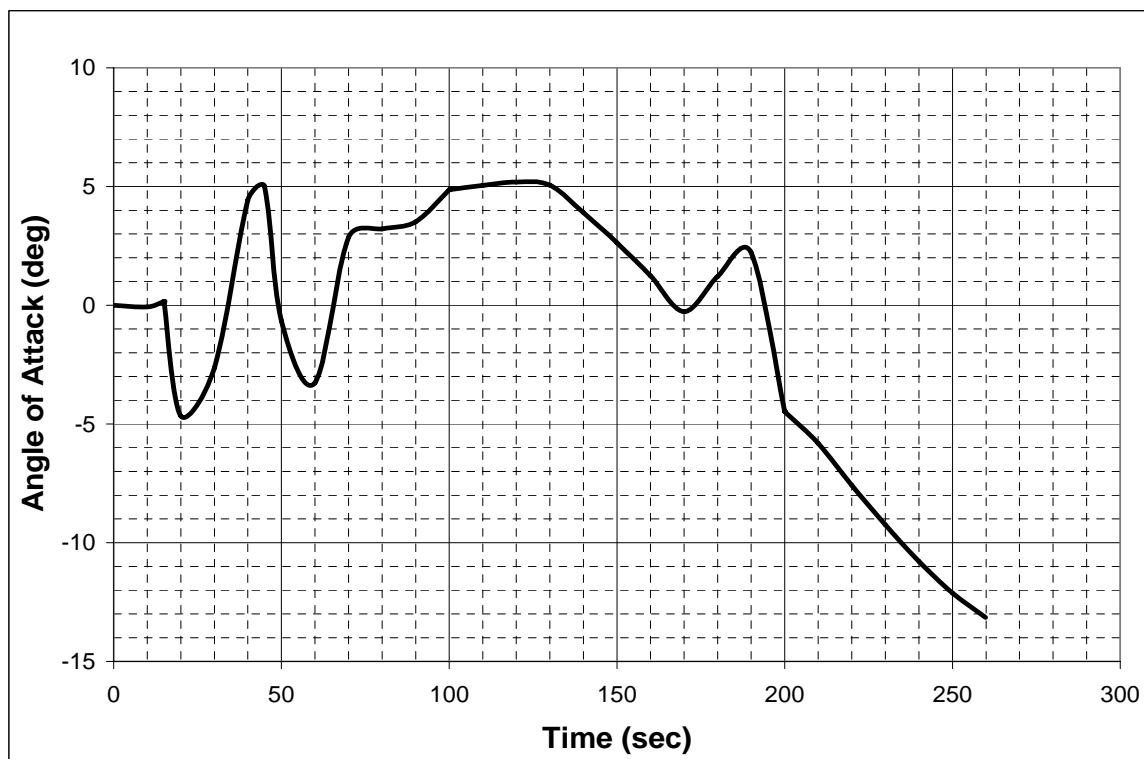


Figure C4. Angle of Attack vs. Time for SSTO Vehicle